REPORT NO. 30-1

EXPLORATION OF THE MOON, THE PLANETS, AND INTERPLANETARY SPACE

EDITED BY ALBERT R. HIBBS

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA
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National Aeronautics and Space Administration
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JET PROPULSION LABORATORY
California Institute of Technology
Pasadena, California
April 30, 1959

PREFACE

Portions of the following report were originated under studies conducted for the Department of Army Ordnance Corps under Contract No. DA-04-495-Ord 18. Such studies are now conducted for the National Aeronautics and Space Administration under Contract No. NASw-6.

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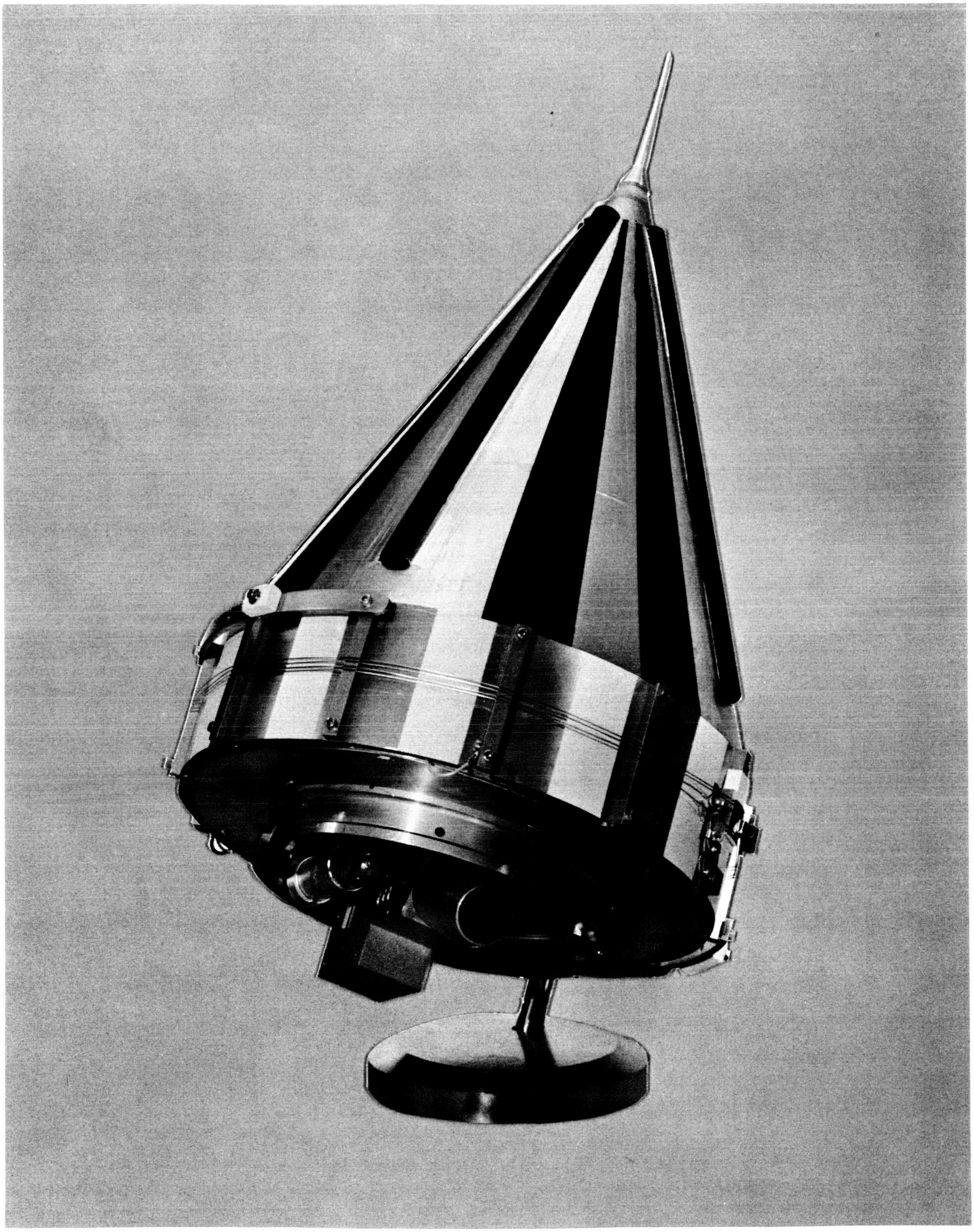
ACKNOWLEDGMENTS

In assessing the technical feasibility of the space-exploration program, the authors of this Report have relied heavily on the broad background and experience of the Jet Propulsion Laboratory. The major portion of our concepts of launching vehicles, payload instrumentation, communications, guidance and control, structural design, tracking, and data processing have come from this source. However, we have also made use of numerous ideas gained from the reports of other organizations or from conversations with their personnel. These have included the Army Ballistic Missile Agency, Convair Astronautics, the Rand Corporation, North American Aviation, Inc., the Space Technology Laboratories, the General Electric Company, the California Research Corporation, and numerous other organizations.

In assessing the scientific and technological merit of various experiments, we have not only relied on the knowledge and skills of the scientists and the engineers at the Jet Propulsion Laboratory, but also have established many rewarding contacts with scientists at universities and other organizations throughout the country. Valuable assistance has been provided by the Space Science Board of the National Academy of Sciences, particularly by the Geochemistry Committee led by Dr. Harold Urey and the Westex Committee headed by Dr. J. Lederberg.

In addition, we have benefited from contact with the National Acronautics and Space Administration and particularly from participation in the NASA Working Group for Lunar Exploration, headed by Dr. Robert Jastrow.

In attempting to estimate the nature of public reaction, we have established contact with various governmental agencies which are concerned with this problem, such as the Central Intelligence Agency and the U. S. Information Agency. We have also conducted numerous informal interviews with friends and neighbors who have no direct connection with this program, aside from helping to pay for it.



**The Jet Propulsion Laboratory's *Pioneer IV* Payload, Launched March 2, 1959—
The First Successful U. S. Space Probe**

ABSTRACT

The Jet Propulsion Laboratory has undertaken a survey of possible objectives in a program of exploration of the moon, the planets, and interplanetary space. This has been combined with a survey of the feasibility of engineering developments which would be required by such an exploration program. The results of this study are presented in this Report.

The Report describes the basis on which the study was conducted, presents a review of current knowledge about the moon, the planets, and interplanetary space, gives a brief summary of the results of the Laboratory study on the feasibility of a program for the exploration of space, describes a program of lunar and interplanetary flights, and outlines the necessary development activities to support the exploration program. The time scale covered extends from 1959 through 1964.

I. INTRODUCTION

In December of 1958, the Jet Propulsion Laboratory was requested by the National Aeronautics and Space Administration to prepare a study of the space exploration program. In particular, the Laboratory was asked to describe those portions of the program wherein it felt it might make the greatest contribution. It was suggested that the period to be covered by the study extend through 1964. Thus, this study was intended to be an outline for a Laboratory program over the next 5 years.

Work on this study was broken down into several different areas; for example, vehicle development, guidance and control, tracking and communications, and so forth. One particular area so defined was the study of the scientific missions which might be undertaken in this program, in particular, those specific scientific objectives which might become the primary objectives of the Laboratory program.

The Laboratory program will consist of both the design, development, and operation of some of the rocket vehicles to be used in the space program and the design, development, and operation of some of the payloads which will carry the scientific measuring devices. In the area of payload development, it is the intention of the Laboratory to concentrate on those payloads designed for lunar and planetary investigations, as contrasted to artificial earth satellites. The area of study of the scientific missions in space, like the other areas of the study program, was governed by this statement of Laboratory intention—concentration on the moon, the planets, and the space between them.

This study of the scientific missions in space had the following objectives: To tie together the important scientific missions with feasible technical developments and produce a realistic 5-year program for the scientific exploration of space. The results of this study are being used in the construction of a Laboratory program for the development of the necessary rocket vehicles and payloads to carry out this program. This Report presents the results of this study of scientific missions. It includes also some portions of other areas in the over-all study program which were used to assess the feasibility of the proposed scientific program.

The results presented herein are not intended to be hard and fast design decisions on vehicles, payloads, or scientific instruments. Furthermore, the schedules presented herein are consistent with scientific potentialities and astronomical dates but are not to be interpreted as program commitments. The results are intended to be as realistic as possible on the basis of present knowledge, but it must be kept in mind that further developments will undoubtedly change many details of the program in a very significant manner. Thus, these results represent a typical program which can be used as a basis for program planning.

II. BASIC PHILOSOPHY

Three criteria have been selected as having the most important effects on the program:

1. Technical feasibility
2. Public reaction
3. Scientific and technical merit

These criteria have been listed, to some extent, in order of relative importance. The question of technical feasibility has been the primary consideration throughout this study. Before one can decide that a particular experiment should be carried out because it is either worthwhile or desirable or both, one must first make sure that it is possible at all.

It is more difficult to determine the relative importance of the last two criteria. Occasionally, they both lead one to the same conclusion. For example, the search for life on another planet is of the greatest scientific importance and, at the same time, is encouraged by a strong public interest. On the other hand, and in this same area, the problem of decontamination of planetary probes may be approached in quite a different manner by scientific and nonscientific groups. Scientific groups recognize the need for decontamination as primary for the success of future explorations for life forms. However, the public may question whether or not it is worthwhile to postpone a Mars shot, for example, for 2 years so that problems of decontamination may be fully solved.

In approaching this particular problem, we have taken the rather optimistic point of view that (1) the problems of decontamination can be successfully worked out in time to meet the proposed schedule, and (2) the public can be educated as to the importance of this problem so that they will neither begrudge the amount of money spent on its solution nor object to the limitation of experiments resulting from its possible lack of solution.

This example is characteristic of the manner in which the basic philosophy of this Report has been applied. An attempt has been made to select the possible, worthwhile, and desirable scientific experiments in the program for the exploration of the moon, the planets, and interplanetary space. A representative flight program has been constructed in which these experiments will be undertaken over the next 5 years. On the basis of the investigations involved in setting up this program, an attempt has been made to select the particular areas of the program which appear to need the most urgent attention.

In applying the criteria of technical feasibility, we have assigned specific missions to the launching vehicles and developed representative estimates of the payload weights which these vehicles can carry to the various objectives. It must be recognized that these weights are far from definite and are intended to be representative of the type of vehicle available at a particular time during the program. Furthermore, we have broken down the payload weights into the various major payload components and have listed representative weights which might be ascribed to each of these components. Here, again, these weight estimates are far from definite, but they are reasonable and serve to point out the problems of weight limitation.

We have also attempted to estimate the degree of complexity which can be assumed for each of these components, including the scientific instrumentation. This estimate has been made with the clear realization that the need for extreme reliability limits the degree of novelty which can be introduced into the various payloads.

In investigating the scientific objectives of the program, we have also considered the purely technological problems which must be solved if the whole program is to be successful. The telemetered information sent back from the payload must contain the results of measurements made for purely engineering objectives. In order to make possible the development of increasingly complex payloads, we must develop a background of engineering design data on the behavior of materials and components in the completely new environment of empty space and the atmospheres of other planets.

The program developed from this basic philosophy and the resulting investigations has been projected 5 years into the future. The validity of estimates of technical feasibility, public reaction, and scientific and technical merit is naturally degraded as we proceed further and further into the future. It was felt that, on the basis of present knowledge, a reasonable prediction could be made for a period of no more than about 5 years.

For this reason, present study does not include consideration of the man-in-space program. It is not reasonable to assume that the man-in-space program would have any direct bearing on the technical problems involved in the exploration of the moon, the planets, and interplanetary space over the next 5-year period. Although many of the scientific and technological experiments which will be carried out during this program will have a definite bearing on the design of the vehicles which carry men to the planets, it is not felt that this objective is in any way inconsistent with the already stated criterion of scientific and technological merit.

Some time after the close of this first 5-year period, the man-in-space program and the interplanetary space program will gradually merge.

Certainly, a manned landing on another planet is one of the most important objectives of a long-range program. Regardless of how clever we become with remote measuring devices, one hard-rock geologist landed on the moon, for example, would be worth many tons of automatic equipment. The public interest in full-color photographs taken by a remote camera on the surface of Mars will be little as compared to the wild reception which will greet the first crew of astronauts which returns alive from that planet.

It is the basic philosophy of this study to develop possible, sensible, and desirable beginning for a program which will eventually take man to the planets.

III. TECHNICAL FEASIBILITY

A. Flight Mechanics

The feasibility of any particular planetary experiment depends upon many factors. Among the most important are (1) the payload which can be carried to the target planet, (2) the accuracy with which an intersection with the target planet can be achieved, and (3) the capacity of the communication system to return to earth the information gathered.

The capacities of a given rocket and payload system in any of these areas can be determined only with the help of an analysis of possible interplanetary trajectories. This analysis must determine the speed with which the payload must be launched from earth, the sensitivity of its final position to errors made in launching position and direction, the possibilities of correcting the course during the coasting period from earth to the target planet, and the distance between earth and the target planet at the time when experimental information is to be relayed back. In order to establish a flight schedule, it is necessary to know the precise times at which planetary experiments can be carried out; an attempt to launch a payload at a particular planet is practicable for only a few days in each synodic period.

An accurate analysis of interplanetary trajectories requires the solution of the *n*-body problem, which in turn requires numerical integration on a high-speed electronic digital computer. However, for many purposes of a preliminary analysis, an approximate scheme is available which permits analytical solutions. This scheme, which might be called the principle of joined conics, makes use of the fact that the various bodies of interest have their most important effect on the flight during different periods of the flight time. Thus, immediately after the payload is launched from earth, the gravitational field of the earth itself dominates the trajectory. After the payload has coasted away from earth, the gravitational effect of the sun takes over as the most important force governing the shape of the trajectory. As the vehicle approaches the target planet, the field of the planet becomes a dominant effect.

The principle of joined conics gives an approximation to interplanetary trajectories in the following way: In the vicinity of the earth, after the burnout of the last stage of the rocket launching system, the trajectory is approximated by a geocentric conic. For interplanetary trajectories, this conic has the shape of a hyperbola. For regions far from earth, the trajectory is approximated by a heliocentric conic which is, in general, an ellipse, at least for the flights considered in this study. In the vicinity of the planets, the trajectory is again a planet-centered hyperbola.

The points at which the various conics are joined together can be chosen somewhat arbitrarily. For this study, the point chosen was at a distance of approximately 2×10^6 km from earth, which is in the region where the effect of the sun's gravity, as compensated by the centrifugal acceleration of a coordinate system moving with the earth, is approximately equal to the effect of the earth's gravity.

A preliminary analysis of interplanetary trajectories has been carried out under the simplifying assumption that all the planets move in circles in a single plane. On this basis, an estimate has been made of the times at which planetary probes can be launched. The results of this analysis are shown in Figs. 1 and 2, wherein firing date is plotted against time of flight for trajectories directed toward Mars and Venus, respectively. The time period covered is from 1959 to 1965; this includes three synodic periods for Mars and four for Venus.

A more exact investigation of trajectory requirements for a Mars probe in 1960 has been completed. This investigation is instead intended to bridge the gap between a simple heliocentric conic analysis restricted to the plane of the ecliptic and a comprehensive conic analysis to be made using the IBM 704 computer, which will soon be available. The present analysis extends the "in-the-plane" analysis to the more realistic three-dimensional case. It also considers the geocentric portion of the flight. One of the most serious limitations of this investigation is that the only heliocentric transfers considered are those in which the earth is at the perihelion of the transfer ellipse, as shown in Fig. 3.

From the earlier ("in-the-plane") analysis a Mars-arrival-date time interval was selected, as shown in Fig. 1. On various days throughout the interval, the coordinates of Mars and the coordinates of earth were obtained from an ephemeris. The geocentric distance of Mars was com-

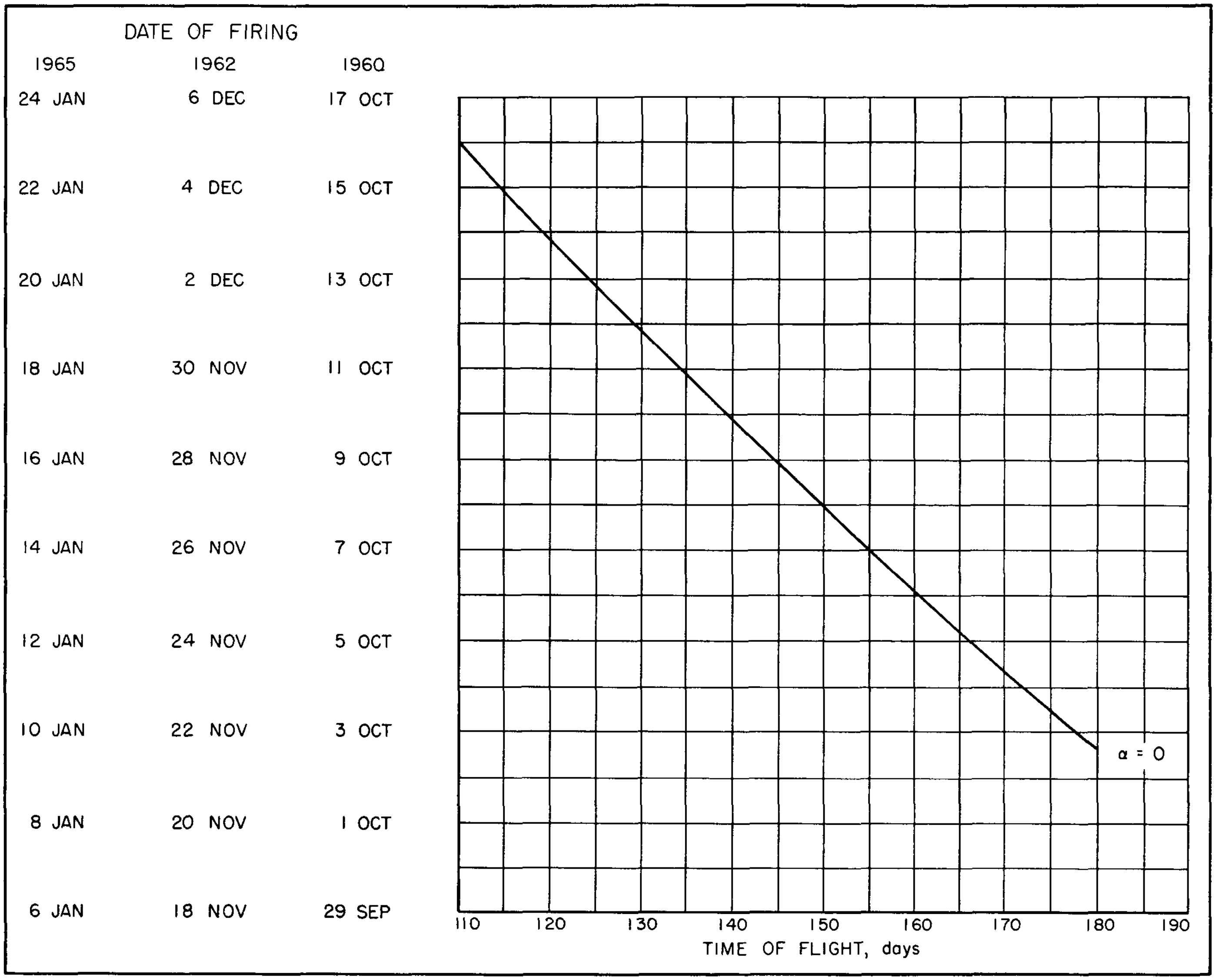


Fig. 1. Approximate Firing Dates for Mars Flights, 1960 Through 1965

puted from the difference in these coordinates. This is the distance over which information must be telemetered from the probe to the earth at the time of arrival. The heliocentric distance of Mars was also computed. The heliocentric geometry is shown in Fig. 4.

Elliptic trajectories were computed with the aid of an electronic computer. It was assumed that the probe entered the heliocentric transfer ellipse at the perihelion of the ellipse. It was further assumed that the heliocentric distance of perihelion was 1 A. U. (astronomical unit: the mean distance from the earth to the sun). For various values of perihelion velocity, time intervals were computed from the perihelion distance of the ellipse to the heliocentric radial distance of Mars at the time of arrival. Also computed was the heliocentric central angle subtended by the heliocentric transfer.

For each value of heliocentric Mars distance, a plot was constructed of transfer time vs central angle. Each value of Mars heliocentric radial distance corresponds to a particular Mars arrival date. A series of heliocentric injection dates was selected, and the ephemeris was consulted for the heliocentric coordinates of earth on these days. The heliocentric angular distance between Mars at the arrival date and each of the positions of the earth at the possible injection dates was computed. Also, the time separation was computed. This curve of time difference

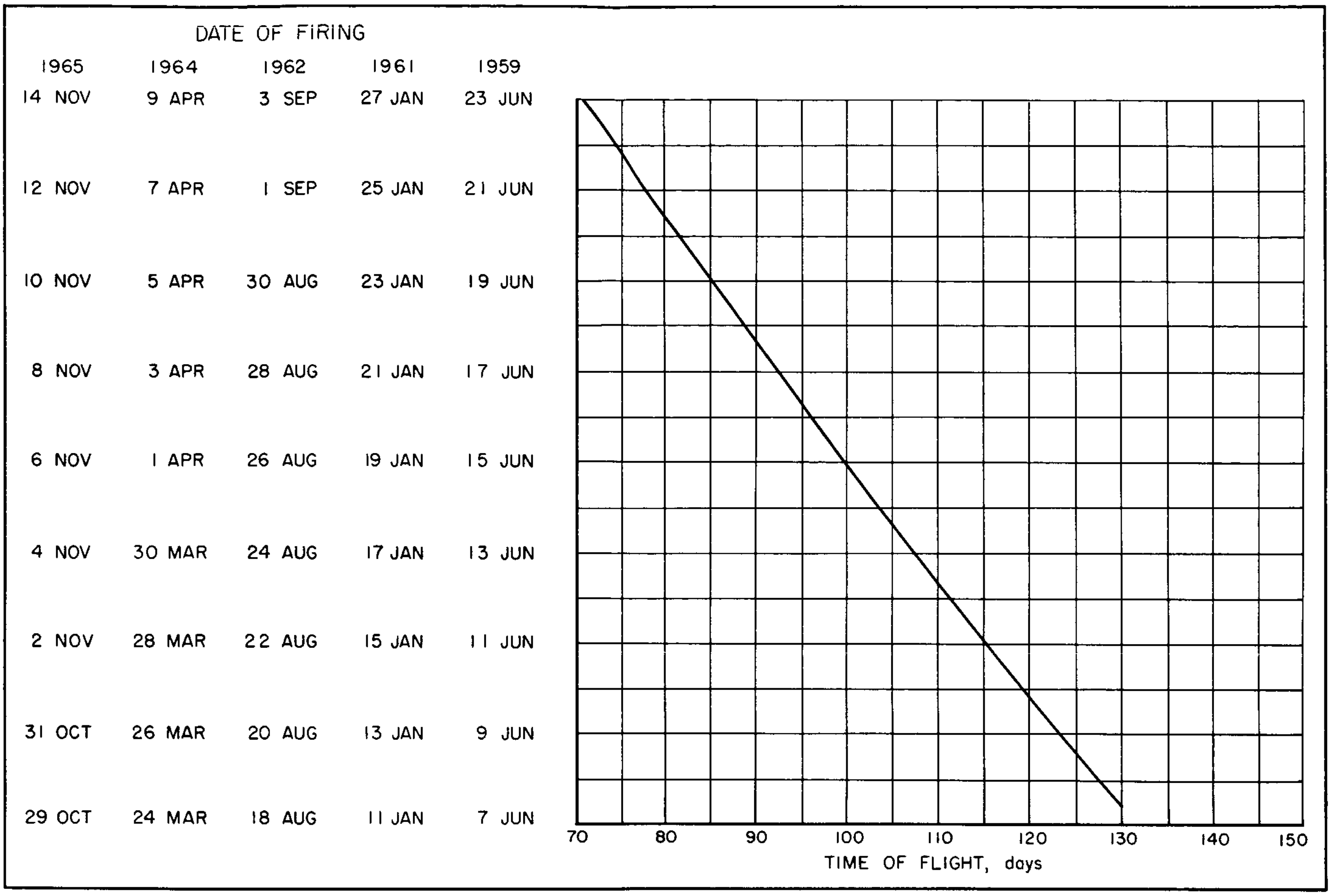


Fig. 2. Approximate Firing Dates for Venus Flights, 1959 Through 1965

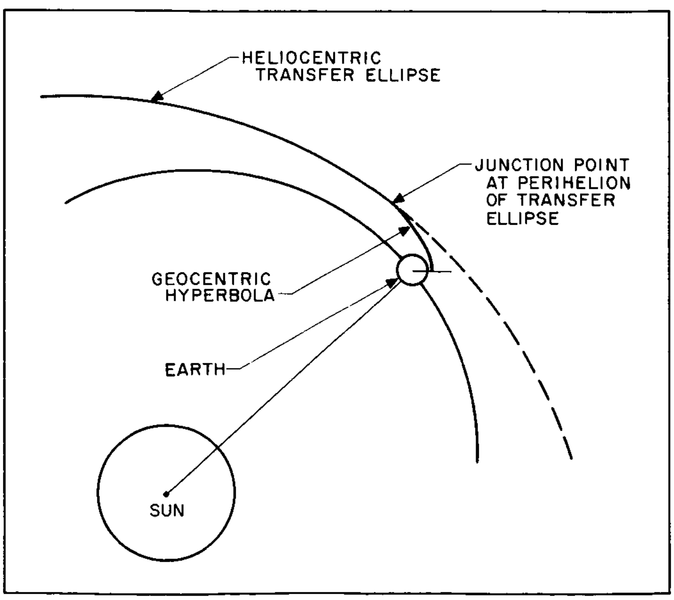


Fig. 3. Diagram of Joined Conics

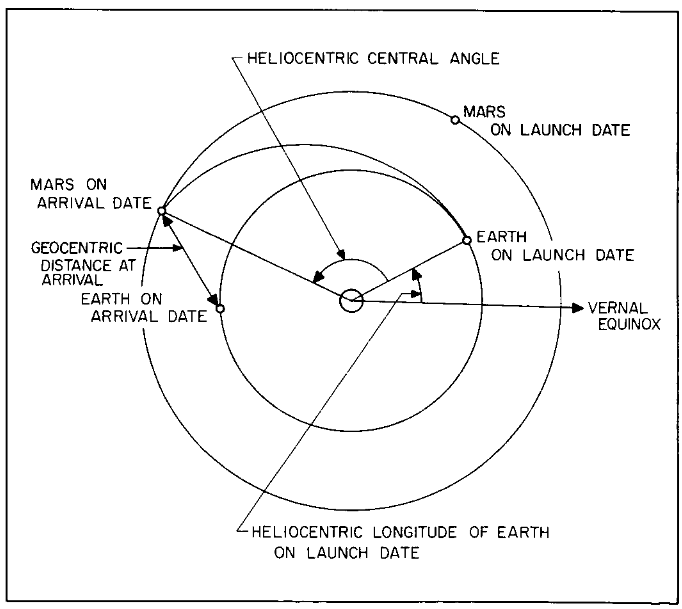


Fig. 4. Heliocentric Geometry for Mars Flight

vs angular separation was plotted on the previously constructed curve from the families of ellipses. The point of intersection indicated the ellipse required and the heliocentric injection date necessary for arrival at Mars when Mars was at the selected heliocentric distance.

By selecting a particular day for the Mars arrival date, the following quantities can be determined: (a) Mars arrival date (Fig. 5), (b) heliocentric distance of Mars at arrival (Fig. 6), (c) heliocentric latitude of Mars at arrival (Fig. 7), (d) heliocentric longitude of Mars at arrival (Fig. 8), (e) heliocentric longitude of earth at arrival (Fig. 9), and (f) geocentric distance of Mars at arrival (Fig. 10).

When the intersection point is determined on the previously described crossplot, the following quantities can be determined: (a) heliocentric central angle (Fig. 11), (b) heliocentric transit time (used as an abscissa in other plots), (c) date of heliocentric injection (Fig. 12), (d) heliocentric inclination of the transfer ellipse (Fig. 13), and (e) heliocentric velocity of the probe at heliocentric injection.

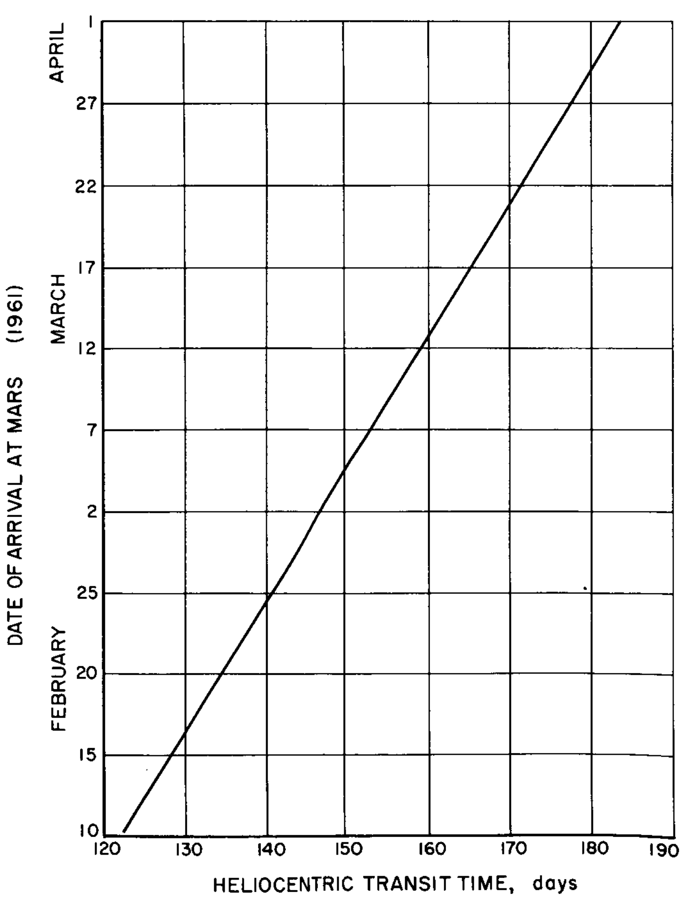


Fig. 5. Mars Arrival Date

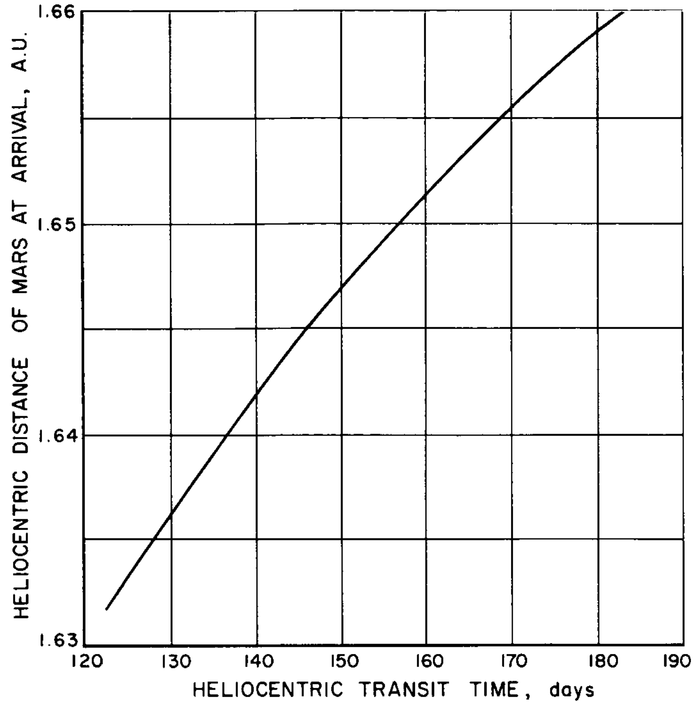


Fig. 6. Heliocentric Distance of Mars at Probe Arrival

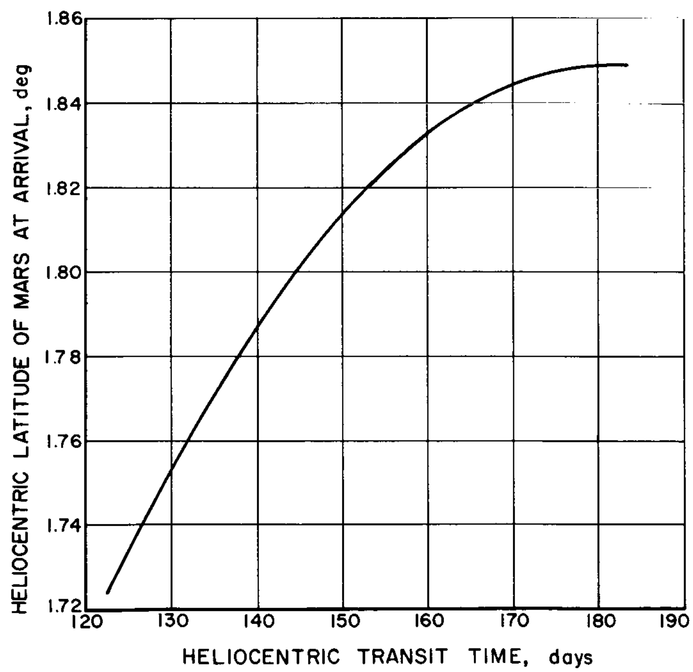


Fig. 7. Heliocentric Latitude of Mars at Probe Arrival

Heliocentric injection is easily analyzed in this investigation since it was assumed that the earth moves at constant speed in the plane of the ecliptic with no radial velocity component. By further simplifying the helio-

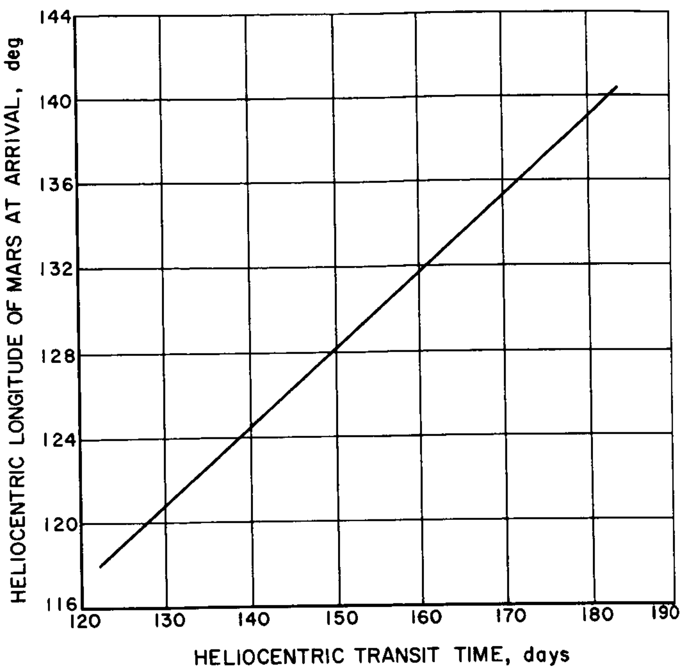


Fig. 8. Heliocentric Longitude of Mars at Probe Arrival

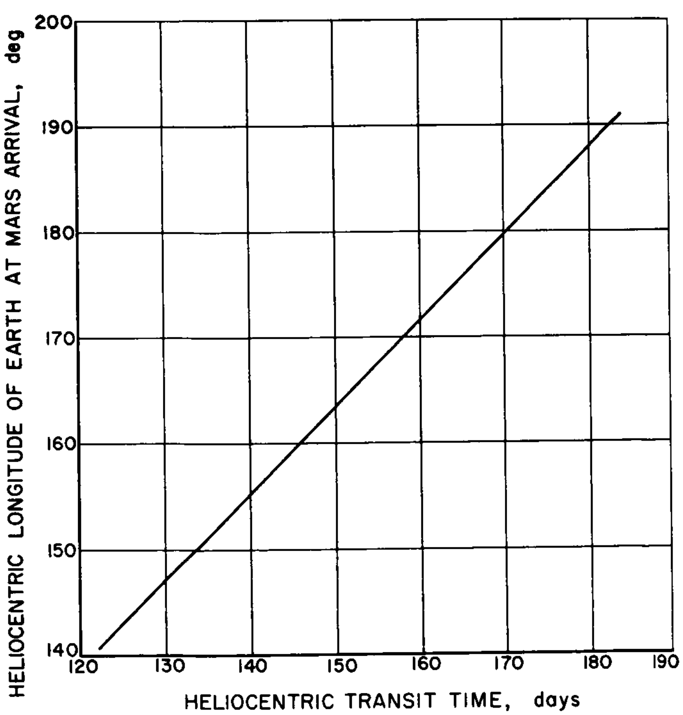


Fig. 9. Heliocentric Longitude of Earth at the Time of Probe Arrival at Mars

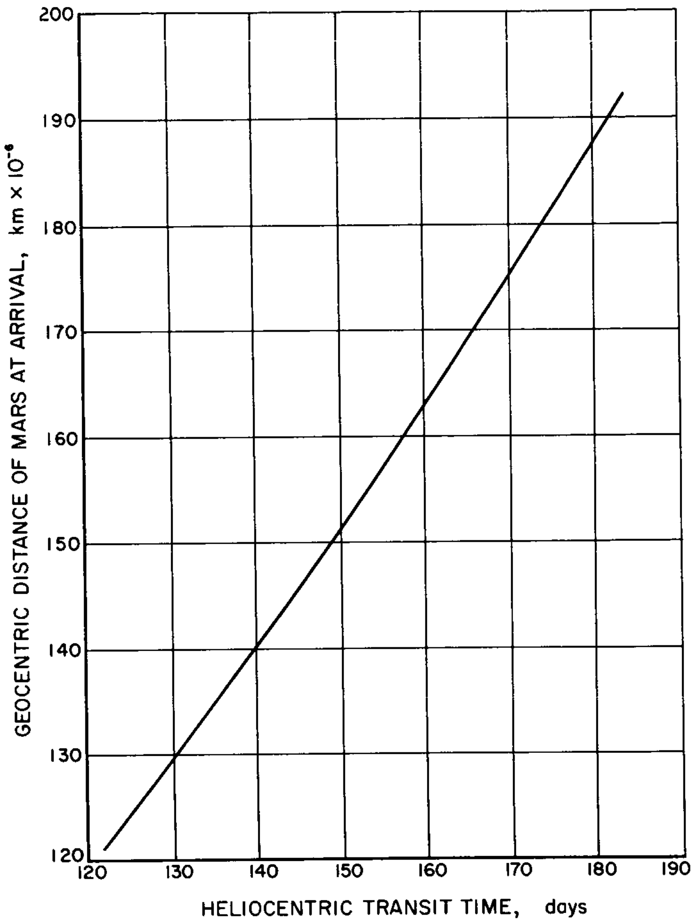


Fig. 10. Geocentric Distance of Mars at Probe Arrival

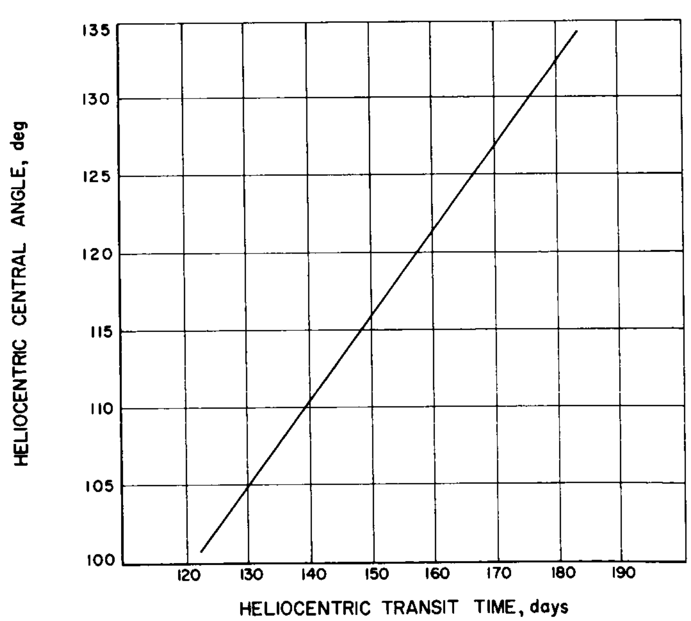


Fig. 11. Heliocentric Central Angle for Mars Trajectory

centric injection, the probe was assumed to have no heliocentric radial velocity component at injection. To clarify the last statement, it must be remembered that the calculations of the time differences and angle differences of transfer are based on the assumption that the probe entered the transfer ellipse at its perihelion. The intersection of time-angle curves required that the perihelion be at zero degrees heliocentric latitude, or in the plane of the ecliptic. In general, the inclination of the transfer ellipse is not zero, and the geocentric distance of heliocentric transfer is not zero (2×10^6 km). Therefore, the probe is neither at the perihelion of the transfer ellipse nor in the plane of the ecliptic at the time of heliocentric transfer. In a rigorous analysis, these factors would be included; in this study they are unimportant.

The difference between the heliocentric velocity of the earth and the required heliocentric velocity of the probe

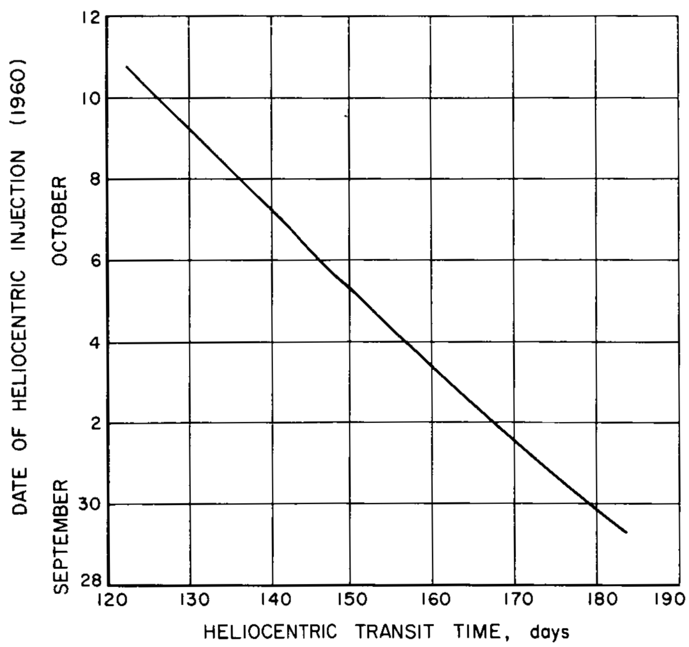


Fig. 12. Date of Heliocentric Injection (Transfer from Geocentric Conic to Heliocentric Conic)

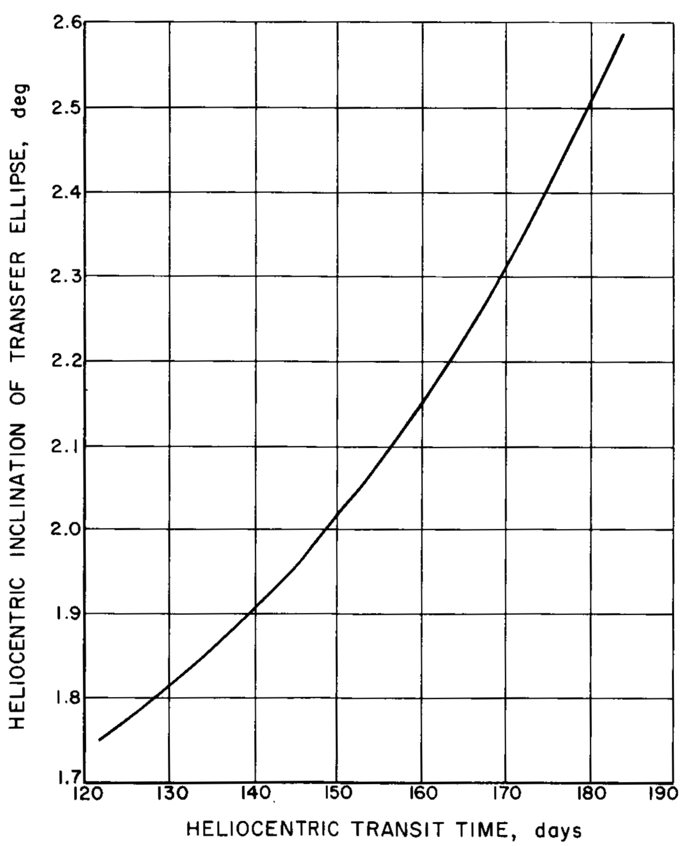


Fig. 13. Heliocentric Inclination of the Transfer Ellipse

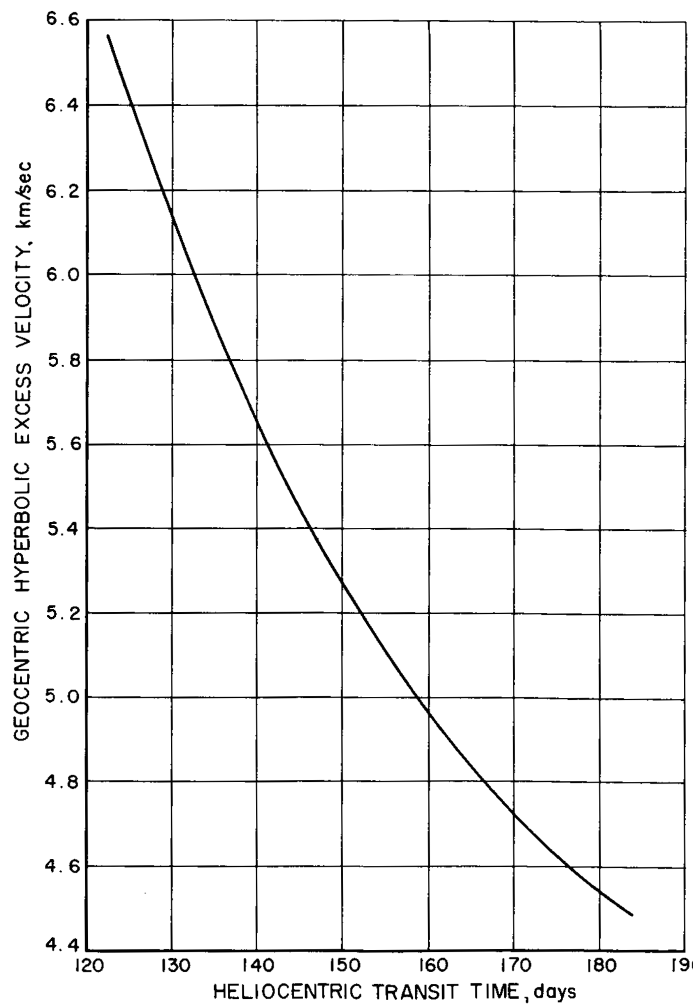


Fig. 14. Geocentric Hyperbolic Excess Velocity

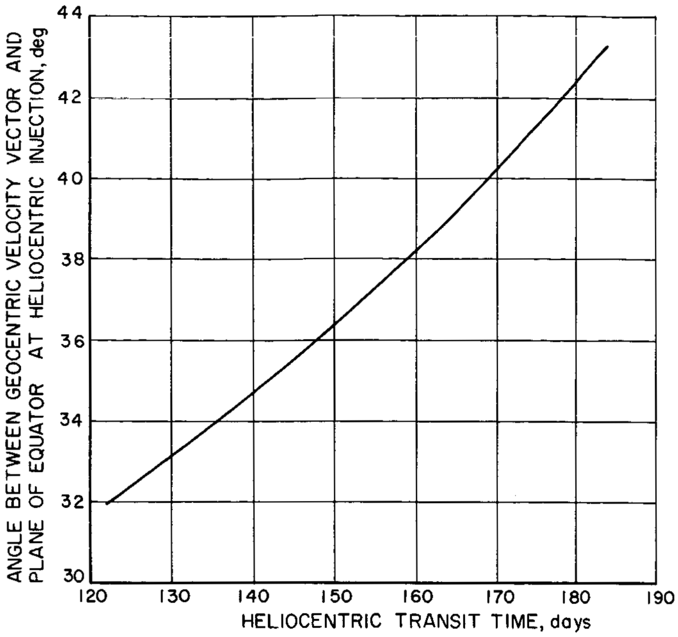


Fig. 15. Angle Between Geocentric Velocity Vector and Ecliptic Plane at Heliocentric Injection

can be computed, both in magnitude and direction. The magnitude is the speed residual of the probe after it has escaped the earth and is called the "geocentric hyperbolic excess velocity" (Fig. 14). To find the angle that the velocity vector makes with the plane of the equator at the time of heliocentric injection (Fig. 15), the angle between the velocity vector (at the time of transfer to the heliocentric conic) and the plane of the ecliptic (Fig. 16), the heliocentric longitude of the earth, and the angle between the plane of the ecliptic and the plane of the equator are used. Since the probe is traveling radially geocentrically at the time of heliocentric injection, this angle is the sublatitude of the probe at heliocentric injection. The geocentric geometry is shown in Fig. 17.

It is next assumed that the probe enters the geocentric hyperbola at the apex at an altitude of 350 km. It is now possible to compute the geocentric transit time; i.e., the time required to go from geocentric injection to heliocentric injection (Fig. 18). It is also possible to compute the geocentric central angle from geocentric injection to heliocentric injection.

A circle is now visualized which is drawn on the surface of the earth in the following way (see Fig. 19): Locate first the subtransfer point, the point on the earth given with the intersection of the surface of a radius

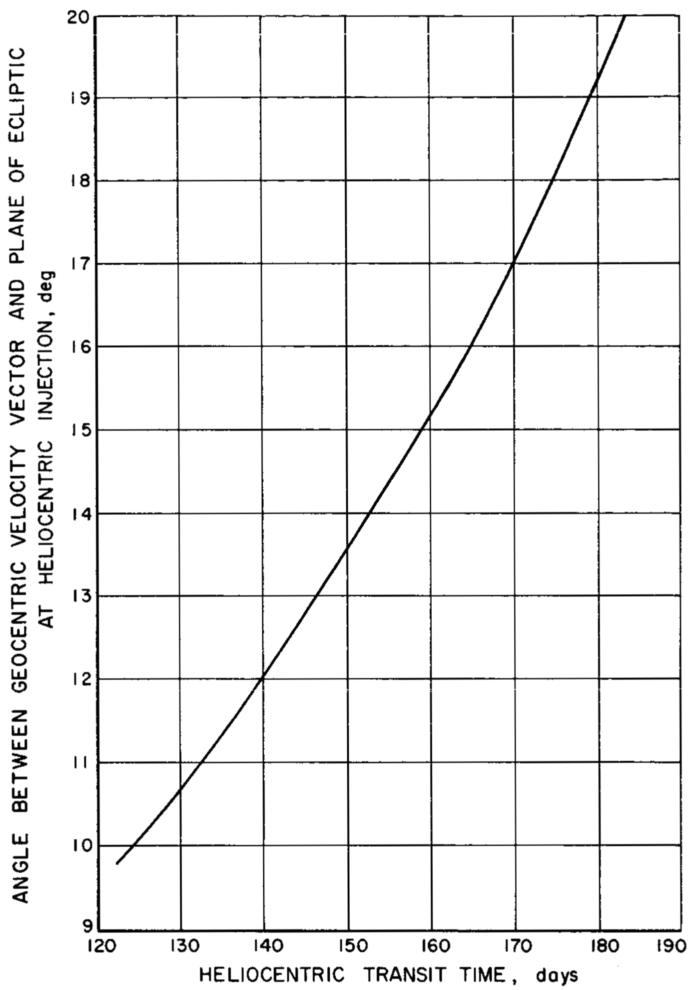


Fig. 16. Angle Between Geocentric Velocity Vector and Equator at Heliocentric Injection

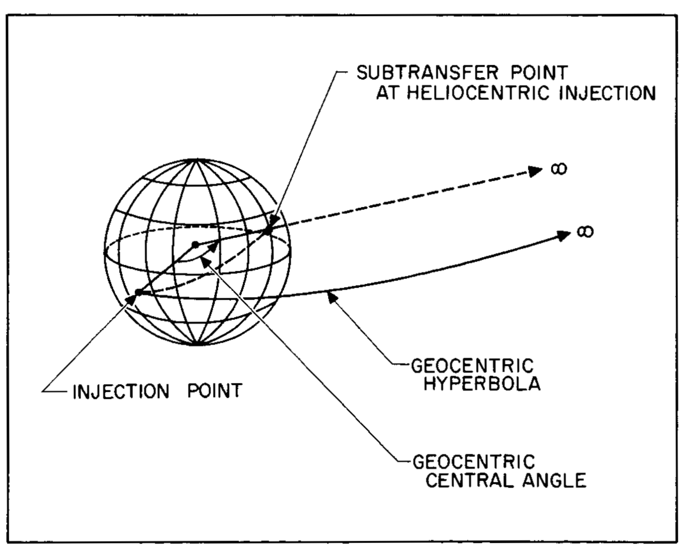


Fig. 17. Geocentric Geometry of Initial Hyperbola

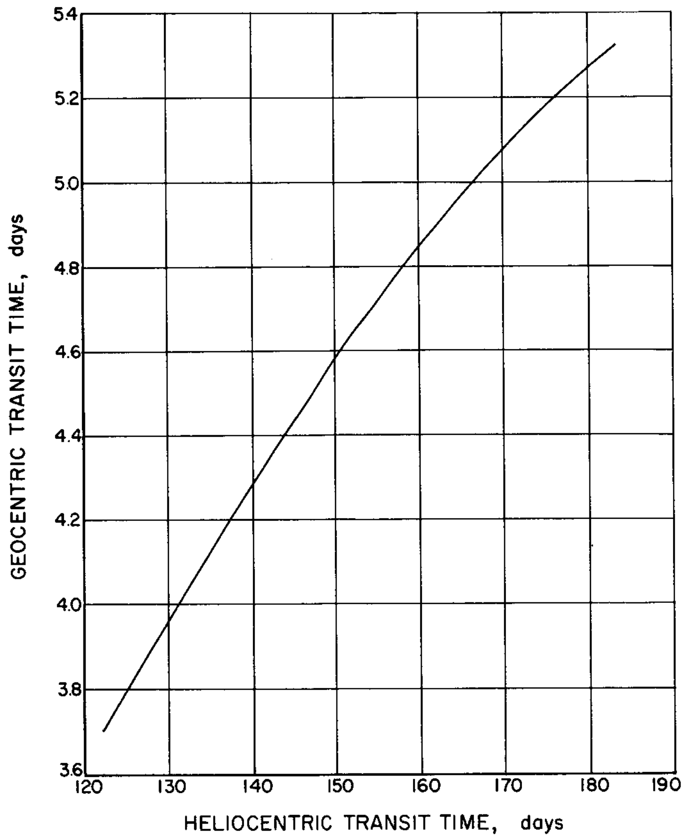


Fig. 18. Time Required from Geocentric Injection to Heliocentric Injection

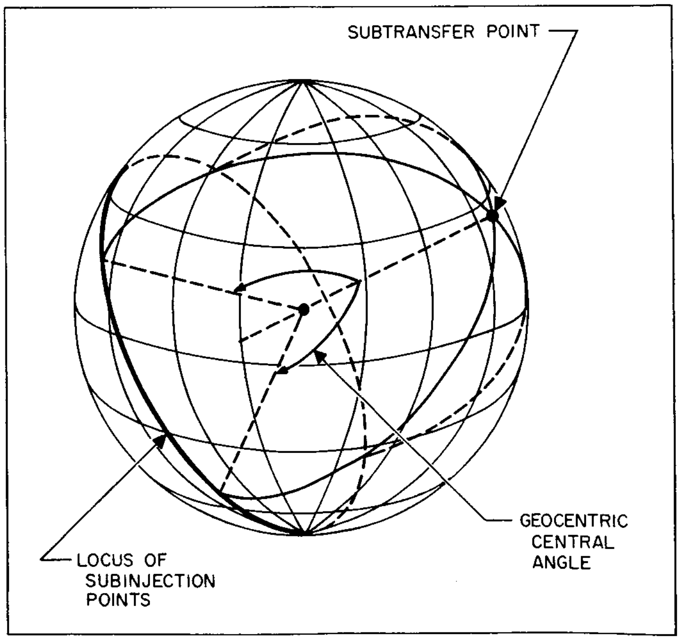


Fig. 19. Locus of Possible Geocentric Injection Points on the Surface of Earth

drawn from the center of the earth outward in the direction of the geocentric velocity vector at the time of heliocentric injection (i.e., transfer to the heliocentric conic). Measure away from that point on the surface through an angle equal to the geocentric central angle. The locus of all points on the surface separated from the subtransfer point by this angle is a circle on the surface. This is the locus of all points over which geocentric injection could be carried out.

In this visualization it is assumed that the earth is not rotating around its axis. Thus, the locus constructed in this manner is actually a locus fixed in geocentric inertial space rather than fixed on the surface. In order to achieve the necessary transfer orbit to Mars in accordance with the geometry used in this study, burnout of the last stage of the launching rocket must occur at a point above this circular locus.

It is also possible to compute the speed which must be achieved at burnout of the last stage in order to obtain the correct hyperbolic excess velocity. The geocentric injection speed is shown in Fig. 20.

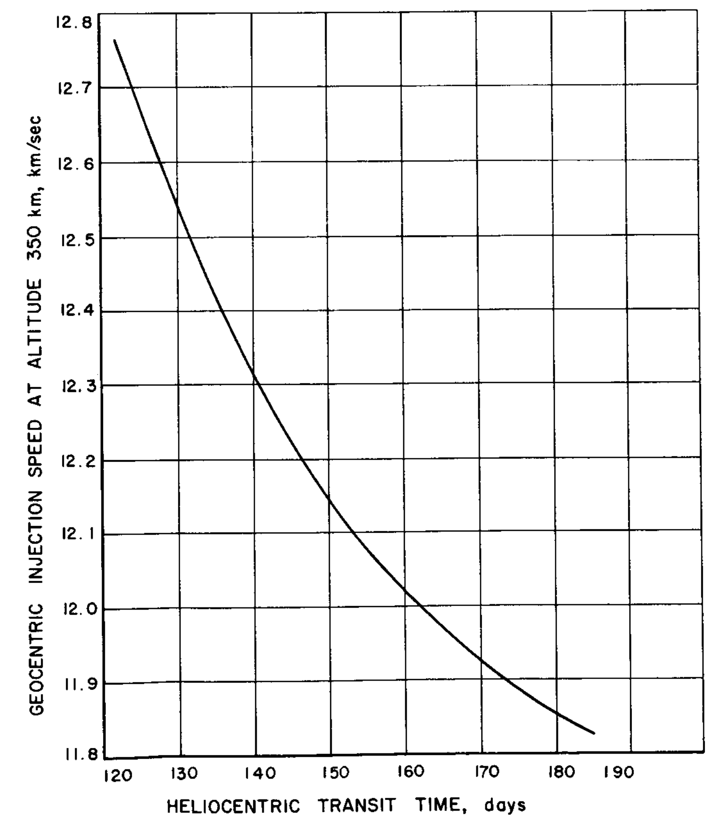


Fig. 20. Geocentric Injection Speed

This analytical treatment of the approximation to interplanetary orbits can be carried through for any planetary target. The results are used to estimate the performance requirements of the rocket, the time of flight, and the distance between the earth and the target planet at the time of arrival.

The results given here apply to a Mars mission in the fall of 1960. Similar results are being obtained for the Venus mission early in 1961 and succeeding flights towards Mars and Venus. The methods used and the types of results obtained are similar to those reported here. A simplified two-dimensional analysis has been carried through for Venus. Results have been obtained giving the geocentric injection speed necessary for a Venus trajectory (Fig. 21) and the distance between the earth and Venus at the time of arrival at Venus (Fig. 22).

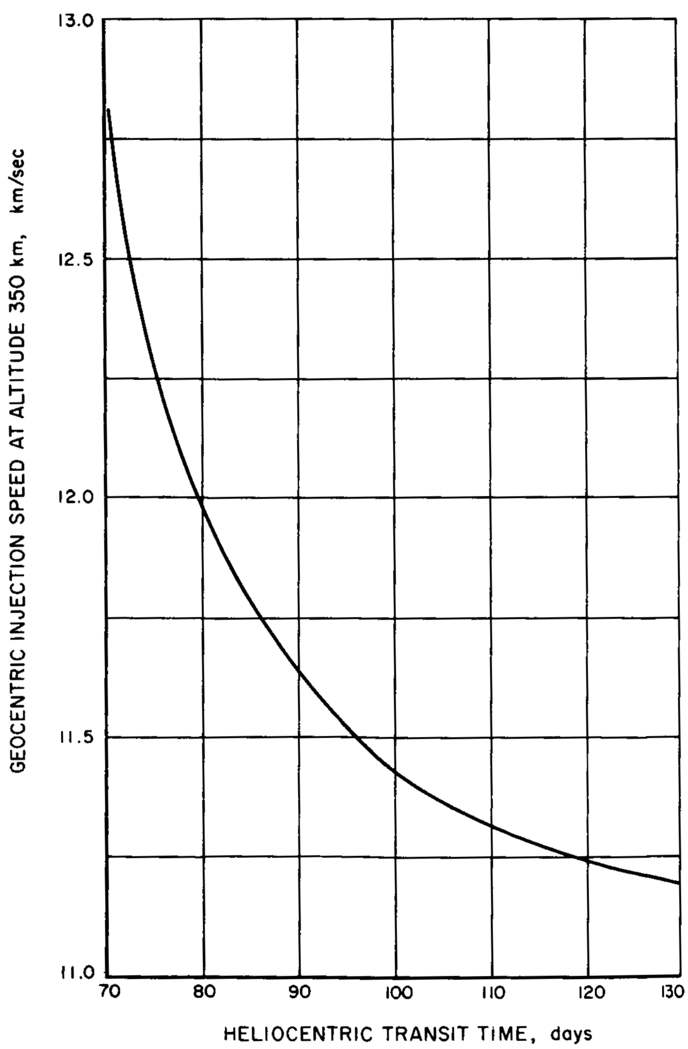


Fig. 21. Geocentric Injection Speed, Venus Trajectory

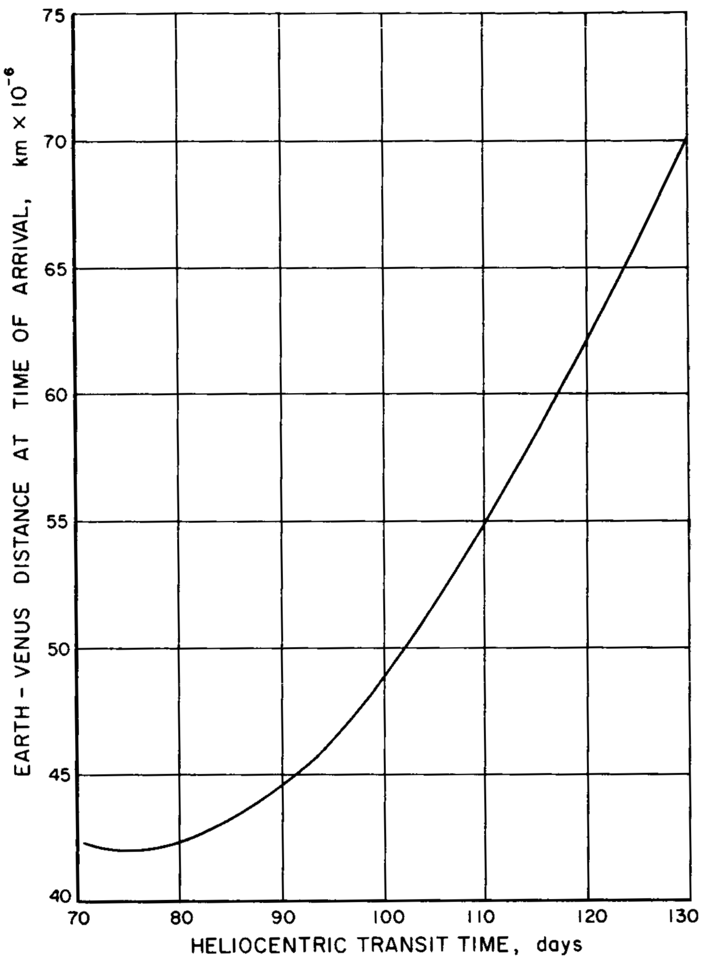


Fig. 22. Geocentric Distance of Venus at Probe Arrival

An additional analysis has been carried out which shows the behavior of the probe in the vicinity of the target planet and, in particular, the following:

(a) the speed in a Mars-centered system at the time the probe arrives at a distance 2 radii out from the center of Mars (Fig. 23);

(b) the speed increment necessary to achieve capture by the gravitational field of Mars when the speed increment is applied at the 2 Mars radii distance (Fig. 24); and

(c) the speed increment necessary to achieve a circular orbit at this distance (Fig. 25).

Similar results (Figs. 26–28) have been obtained for the behavior of the payload in the vicinity of Venus. These results are necessary in order to compute the size of the retro-rockets necessary to establish an artificial satellite around the target planet.

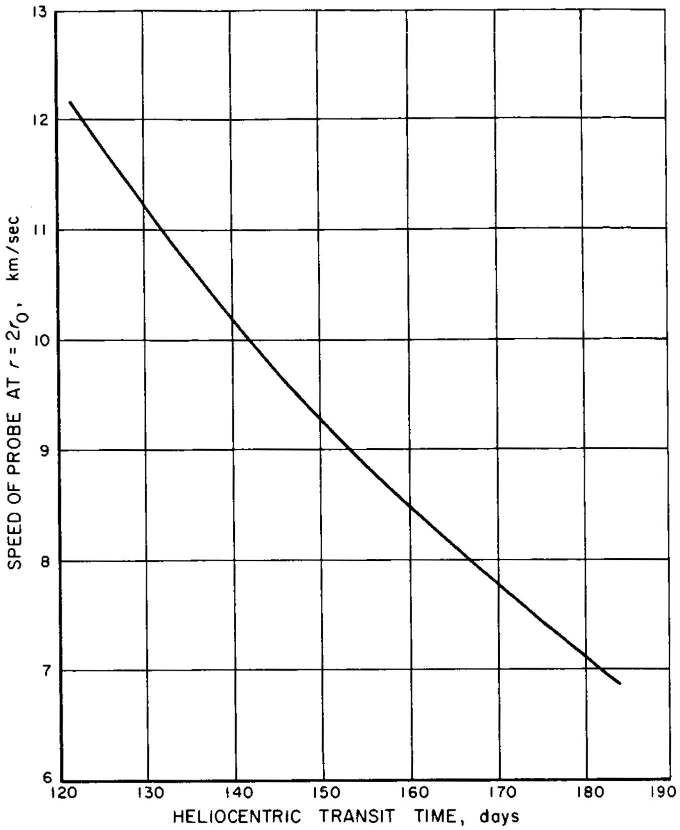


Fig. 23. Speed Relative to Mars of Probe at 2 Radii from Center

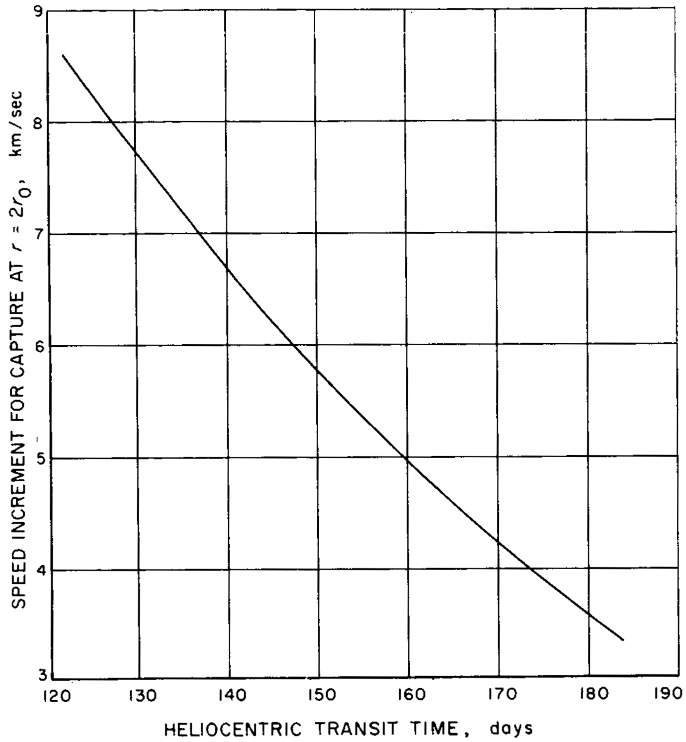


Fig. 24. Speed Increment Required for Mars Capture at 2 Radii from Center

It is also possible to estimate the sensitivity of miss distance at the target planet to errors made during the launching phase. These numbers depend critically upon the shape of the trajectory; for the Mars trajectory in general, they have the following values: An error of 1 deg in the direction of the geocentric velocity vector will contribute between a few hundred thousand and 2×10^6 km to the miss distance. An error of 1 meter/sec in the speed of geocentric injection will contribute between 10,000 and 50,000 km to the miss distance at Mars (see Fig. 29).

B. Vehicle Configuration

The most important factors in determining the technical feasibility of a space mission are the performance and reliability of the rocketry system carrying the payload. The mission schedule presented herein is designed to fit into the National Space Vehicle Program (Fig. 30).

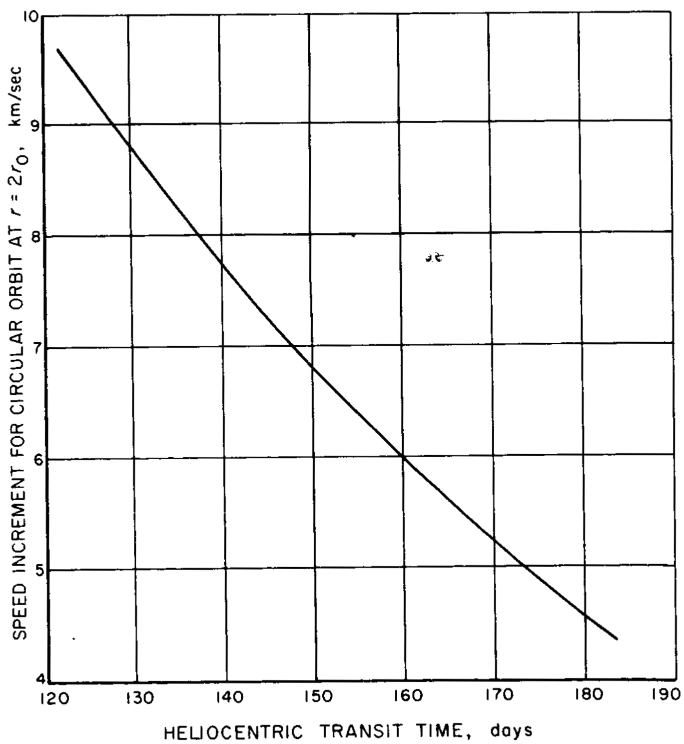


Fig. 25. Speed Increment Required for Circular Mars Orbit at 2 Radii from Center

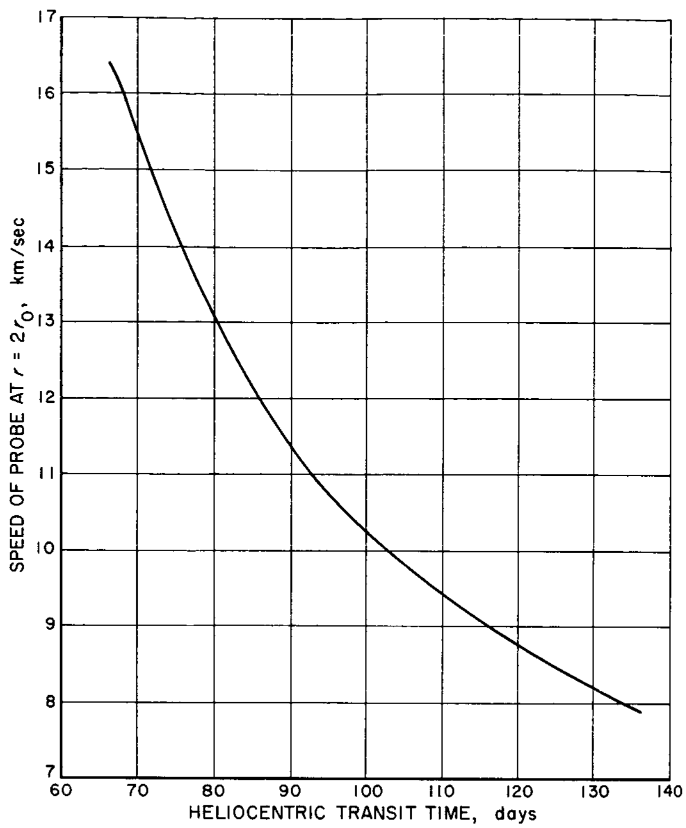


Fig. 26. Speed Relative to Venus of Probe at 2 Radii from Center

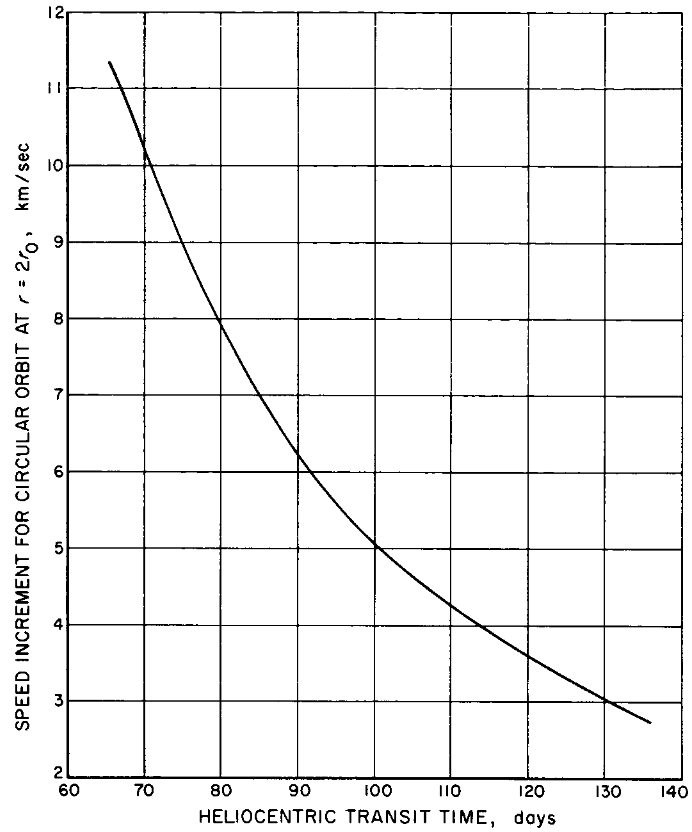


Fig. 28. Speed Increment Required for Circular Venus Orbit at 2 Radii from Center

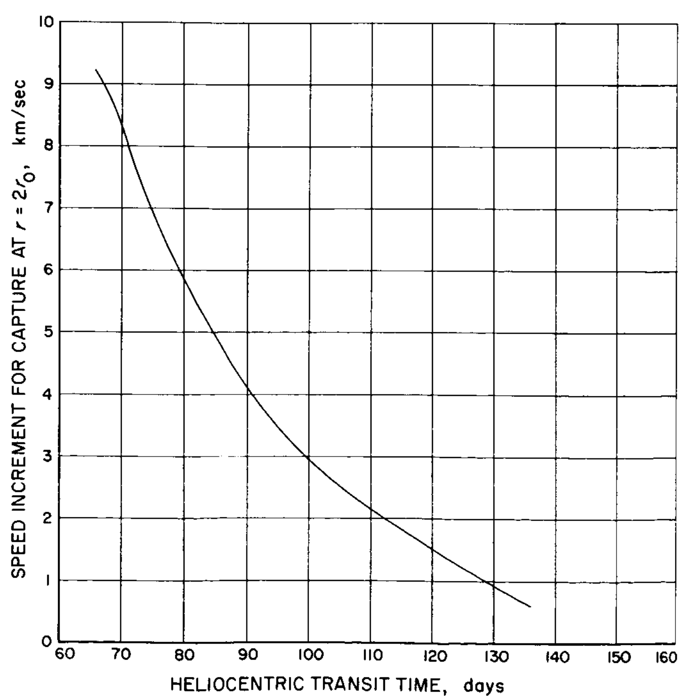


Fig. 27. Speed Increment Required for Venus Capture at 2 Radii from Center

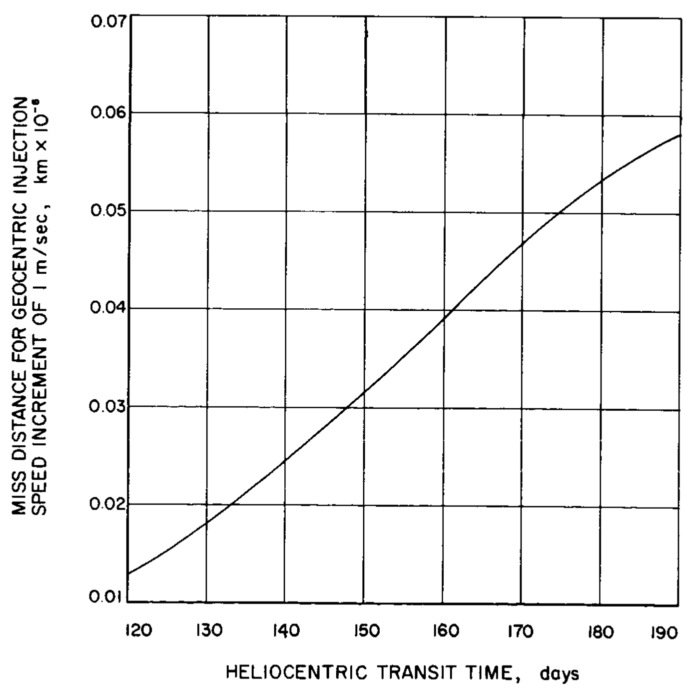


Fig. 29. Error Coefficient for Variation in Geocentric Injection Speed, Mars Trajectory

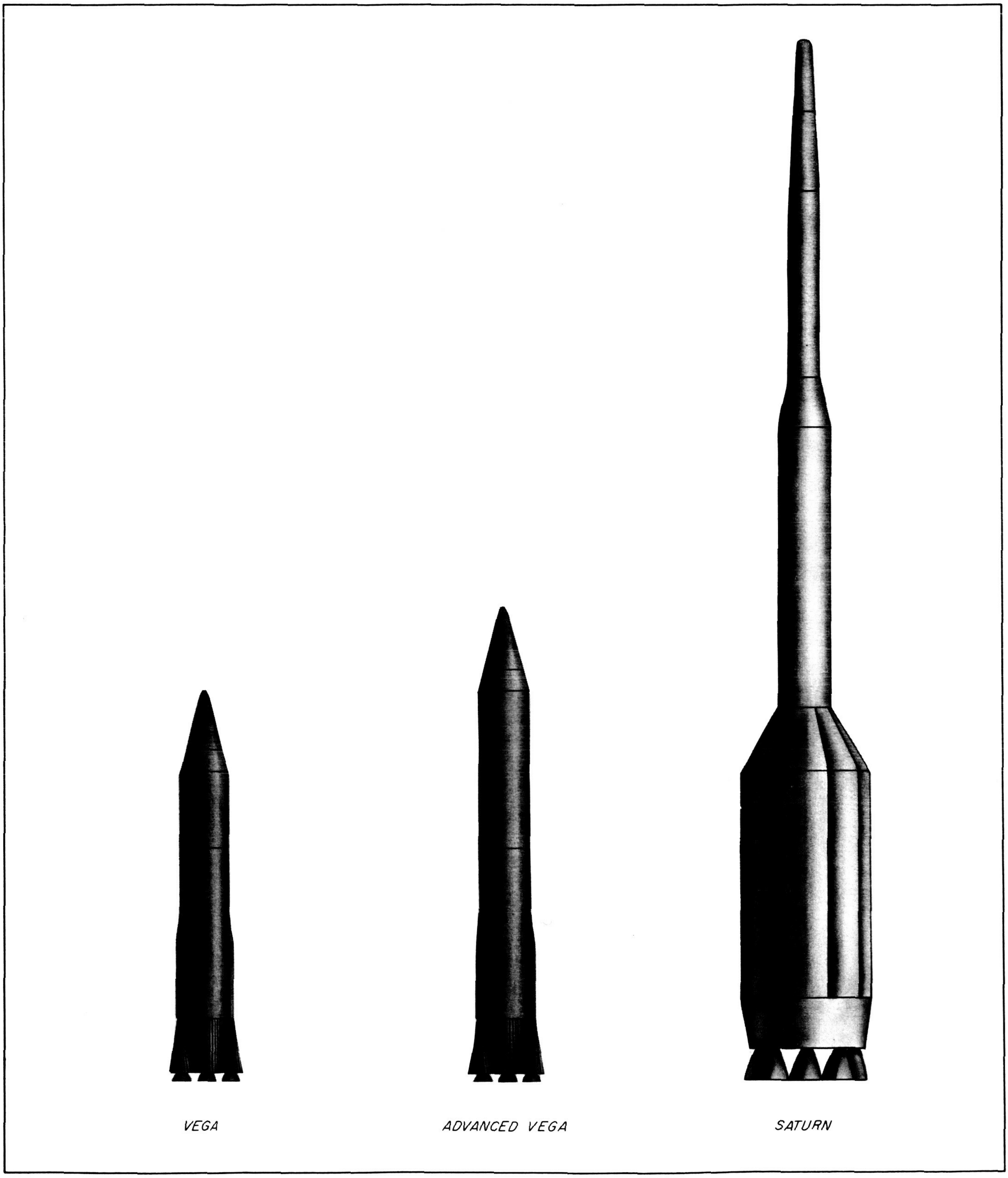


Fig. 30. Typical Space Vehicles of the National Program

The first general-purpose vehicle of the national series is the *Vega*. It is in the generation of space vehicles succeeding the IRBM series and, for the next 2 to 4 years, will serve as the vehicle for most of the missions.

The first stage of the *Vega* will be an *Atlas* ICBM structurally modified to support two upper stages. The second stage will be a modified G.E. 405 rocket, and the third stage will be constructed around a JPL 6000-lb-thrust rocket engine. The latter is now under development at the Jet Propulsion Laboratory.

The JPL 6K engine will utilize storable propellants, nitric acid and hydrazine. Another interesting feature of this rocket engine is that eventually it will have the capability of being re-ignited.

The first of the *Vega* vehicles may be ready by the middle of 1960; in the subsequent 2 years it could be used for 5 different space missions. In the fall of 1962, an advanced version of the *Vega* vehicle may be ready for orbiting a satellite around Venus. The *Advanced Vega* will incorporate a higher energy propulsion system in its second stage. For example, this stage might employ the hydrogen–oxygen propulsion system called *Centaur* now under development.

By 1963, the first of the *Saturn* vehicles may be ready. The *Saturn* is the third generation of the proposed rocket vehicles in the National Space Vehicle Program. It will consist of a cluster of eight *Jupiter* IRBM's generating a thrust of approximately 1,500,000 lb. This vehicle is being developed by the Army Ballistic Missile Agency and will be employed in deep-space missions through 1964 and beyond.

The fundamental characteristics of the vehicles proposed for the next 5-year period are listed in Table 1. In all cases, at least three stages will be used for those missions involving exploration of the moon, the planets, and interplanetary space.

Guidance and attitude-control systems will be developed to control the trajectories of these vehicles during the launch phase. These systems are called "injection guidance." The development of these systems will rely upon developments which have already been undertaken in guidance systems for military rockets.

Because of the extended range covered by the vehicle during its burning phase and because of the frequent necessity of coasting in a "parking orbit" before ignition of the final stage, it appears that, at least for the upper stages, inertial guidance is more desirable than radio guidance.

The schedule for the development of the guidance system is similar to the schedule for the development of the vehicles. The first few vehicles will be guided by systems that are only slightly modified from those already available. As the program develops, lighter-weight guidance systems designed specifically for the space vehicle application will be introduced.

It is difficult to make an exact estimate of the payload weights which will be carried by any of these vehicles on the various missions. The best that can be done at the present time is to make a rough estimate which should be good to within at least a factor of 2 for the payloads that can actually be carried. The gross payload weights which have been assigned to various missions are listed in Table 1. In general, the *Vega* will carry a payload of approximately 400 to 700 lb to the nearer planets. The *Advanced Vega* will carry a payload of between 700 to 1400 and 2000 lb to the nearer planets; whereas the *Saturn* will carry a payload of between 4000 and 10,000 lb.

Table 1. Fundamental Characteristics of Proposed Vehicles

Vehicle	Propellant		
	Stage 1	Stage 2	Stage 3
Vega	LOX–RP-1	LOX–RP-1	storable N_2O_4–N_2H_4
Advanced Vega	LOX–RP-1	LOX–H_2 (*l*)	storable N_2O_4–N_2H_4
Saturn	LOX–RP-1	LOX–RP-1	LOX–RP-1

LOX: liquid oxygen, RP–1: kerosene, H_2 (*l*): liquid hydrogen.

The dates assigned for use of the vehicles do not, in general, begin with the date at which the particular vehicle is scheduled for its first flight test. Only the *Vega* will be employed as a space vehicle on its first test. It is assumed that the application of the *Advanced Vega* and the *Saturn* to the space exploration program will follow some months after the first flight tests of these vehicles.

C. Instrumentation

1. Photography. Of the instrumentation proposed for the missions in the NASA study, photographic equipment is probably the most important. From an astronomical standpoint quite a number of questions can be resolved by adequate photographic data, particularly if it becomes possible to take pictures in very selected portions of the spectrum, e.g., visible light, ultraviolet and infrared.

Aside from weight constraints, the most difficult problems in utilizing photography in deep-space exploration are the following:

1. Because of the presence of the Van Allen belts, taking film away from the vicinity of the earth without appreciable radiation fogging is difficult.

2. Because of the limited communication bandwidth available at planetary distances, it will be somewhat of a problem to transmit the pictures. It must be realized that, since even crude pictures contain the order of 10^6 bits of information which must be telemetered back with an available information rate sometimes as low as 10 bits per second, it will take a little more than 20 hours to do. Naturally the total number of bits in a picture is tied to the definition or raster required as well as the number of shades of gray, and a degradation of these requirements will make the problem somewhat easier.

3. The accuracy with which the attitude of a vehicle can be controlled enters into the problem of pointing the camera. If the film employed is somewhat insensitive to radiation, it will be comparatively slow. It then becomes necessary to maintain this attitude with great precision over an appreciable time.

The present state of the art makes it possible to take pictures out in space employing a raster of 200×200 lines per in., with a line resolution along the line of 512 lines per in. The capability of the ground system is 1200×1200 lines per in. This resolution is quite far from that which can be expected in the years to come. It is anticipated that, by continuing the development of the present system, it can be improved by a factor of 2 within a year. The present camera is able to take 6 pictures, develop them and telemeter them back at a rate of 1 picture per hour. The actual effect of radiation in the belts upon the performance of this system is yet to be found, and the best procedure will probably be to expose some film to a known test pattern, transport it through the radiation belt, and then develop and telemeter it back.

There are a number of ways to deal with these problems, and various ways are under consideration to circumvent them. Regarding the radiation-belt problem, it should be possible to develop a film with an increased dynamic range to limit the influence of radiation. An alternative is to use an electronic film based on solid-state phenomena; and research should be accelerated in this direction. This would simplify the procedure of developing the film, which with normal film is not easy to accomplish on an automated basis out in space.

The alternative to picture-taking is television; a number of systems have been suggested, and some of them have been carried through partial development. One is a vidicon system which allows a picture to be taken and then stored electronically. The scanning rate of this picture is then regulated to conform to the available information rate. The resolving power and weight are still problems to be solved. At the present time, it is possible to obtain vidicon tubes with a sensitivity in the red region of the spectrum; development has been started to extend the sensitive region of these tubes to the far infrared. Development of high-resolution television has been started in industry. These systems have rasters of 2500×2500 lines per in., with spectral sensitivities that can be chosen in ranges from the ultraviolet to the far infrared.

The communications problem may be solved by proper coding of the telemetering. It is not necessary to telemeter the information on an absolute basis; it is possible to telemeter the changes in information from the previously sent information and thereby make it possible to send more information with a given bandwidth. All in all, a number of promising possibilities exist in the field of space photography and/or television, but considerable development will be necessary before high-grade photography can be accomplished.

2. *Magnetometers.* Fluxgate or nuclear-precession magnetometers for space exploration are already available and more refined instruments with greater sensitivity and reliability will be ready for space flight within 6 months from now. Among the types that appear most promising are the alkaline-vapor magnetic-resonance type, with a sensitivity of 10^{-4} gauss, and the nuclear precession-type recently improved by Russian scientists. Magnetometers of an extremely rugged construction capable of withstanding landing shocks in the vicinity of 1500 g are under development at the present time. These magnetometers will permit placing stationary instruments on the moon prior to the soft-landing phase of the program.

3. *Cosmic-ray instrumentation.* The number of experiments proposed in this particular field is very great and of considerable interest to both cosmologists and astrophysicists. The instrumentation will require considerable development of individual components and transducers and further system development is needed.

In the first category, a few items slated for development, or rather redevelopment, should be mentioned: (1) scintillation detectors with adequate sensitivity and sufficiently low noise, (2) sensitive photomultipliers rugged enough for the boost phase, (3) transistorized pulse-height analyzers with low power requirements, good reliability, and very good discrimination, and (4) ruggedized ionization chambers and counters for rough-landing vehicles.

The second category (system development) includes such things as system engineering, packaging, and optimizing the data received in comparison to the power and weight penalties implied by the instrumentation. A great number of these experiments have already been done on earth and must now be performed out in space. Development of the instrumentation for these experiments seems to be progressing reasonably, but a certain amount of re-engineering will have to be done, particularly in the field of automation and self-calibration, in order to render a maximum of data for the weight and power expended.

4. *Meteor detectors.* The experiments flown to date have shown that the concentration of micrometeorites is very nearly that which was originally anticipated. The gauges that have been employed in this field have been very primitive, giving data only on a go/no-go basis. For a complete study of interplanetary material it is necessary to measure not only the abundance but also the total momentum of micrometeorites and also, if possible, their direction. This will necessitate development of transducers of a more complicated nature. Gauges based on the established leak rate of spheres of appreciable size would be able to indicate the number of puncturing impacts per unit area and time, and gauges based on secondary emission would give the total momentum. This will take considerable development, but it will take still more engineering to construct simulation devices to test these gauges before they are sent aloft. The problem of meteor detection, which will be important in flights to come, must be solved now.

5. *Mass spectrographs.* A number of experiments using mass spectrographs have already been made in the field of upper-air research. These instruments have been carried by rockets and have obtained some data; however, considerable development work remains to be done if these instruments are to be able to analyze the composition of gases in the neighborhood or on the surface of a planet. Conventional magnetic-deflection instruments are competitive in almost every way with the best rf instruments being used in rockets. They do have the advantage of having been long tested in precise gas analysis so that they give more reliable results, and research and development along these lines should be stimulated and supported in addition to continued experimentation with rf instruments. With available instruments of both types, complications arise in measuring the neutral matter in space because of the very low pressures that exist there. The lowest pressure at which a magnetic-deflection instrument will give an acceptable current lies around 10^{-11} mm Hg and the corresponding figure for an instrument of the rf type is 10^{-10} mm Hg. These limiting pressures are far too high; it becomes necessary to use a matter accumulator to collect material for a long period of time and then release it into a mass spectrometer for analysis. The use of titanium as a matter-collection device has been suggested, but considerable work must be done to understand the basic mechanism of this metal as well as to investigate the reversibility of the reaction. It might very well be that titanium will not be used, but preliminary study shows it has possibilities.

For the electronics used in conjunction with these instruments, some development will be necessary, mainly because of the power and weight constraints associated with the payload, as well as the problems of outgassing the payload to prevent gasses from that source from interfering with the measurements.

When these instruments are used to analyze the composition of an unknown atmosphere, a further problem arises in regulating the pressure. The output figure is critically dependent on the maintenance of pressure at a known constant level. While all these problems are by no means insurmountable, much development will be needed before reliable figures can be derived from this type of instrumentation.

6. Ion probe. Determination of the properties of the corpuscular radiation from the sun is a scientific mission of great interest. Instrumentation to measure the character and intensity of this radiation must be developed. A modified Langmuir probe or an improved model of the probe used in *Sputnik III* could be used. In all cases, this type of instrumentation will need electrometer tubes rugged enough to sustain the boost phase and at the same time much less sensitive to emission variations derived from variation in the filament supply employed. In the same category are equipments to measure the direction, energy spectrum, polarity, and flux of these particles. These measurements are important not only from an astrophysical standpoint but also because of the expected contribution of these particles to the solar pressure on objects of reasonable area, such as antennas or solar panels. This pressure will have to be taken into account in the guidance and attitude control of space vehicles.

7. Spectrophotometers. These instruments have been invaluable tools of astronomers on the earth and with the advent of deep-space exploration more elaborate requirements for this type of instrumentation will emerge. Far above our atmosphere, with an undistorted view of the objects in question, these instruments will be called to work in the extreme ends of the spectral range. A broad development program will be necessary to adapt these instruments to space flight. Some preliminary work along these lines has been started, but a more more general program is required. Consider the problem of assigning a spectrophotometer to work in the infrared portion of the spectrum. Prism materials that will be able to stand the vibration and acceleration of the boost phase do not now exist. A broad investigation of the materials question will be required, as well as development of techniques of automation and self-alignment for these instruments. Employment of such instruments for space exploration will also, in most cases, put severe requirements on the attitude control of the payload, since for the normal case the cone angle to the target will be very small. If this question can be solved and the angle maintained for reasonably long periods of time, requirements on the information rate needed to telemeter the results should not be too severe. Development of sensors for aiming and tracking these instruments will be necessary, since the information from the spectrophotometer will, by its very nature, be difficult to use as the criterion for correct alignment to the target.

8. Timers. In addition to the timing requirements in the boost phase, the payload itself will also have certain timing requirements. The timing functions required of these timers include cycling the transmitter, switching telemeter channels when the number of experiments in the payload exceeds the number of channels available, acting as general logic circuitry in the payload, etc. The timers employed must, by the nature of the mission, have a long life. Associated with logic circuitry, they should operate in the payload with their own independent power supply. A reasonable amount of development work will have to be performed for this timing problem in all its detail.

The projected timer will have to be a solid-state type because of the power constraint, and should be able to be programmed easily, both as a one-shot timer and as a recurrent one. The over-all timing cycle will depend on the mission; the timer should be able to function for at least 6 months. The over-all power drain should be no more than 200 mw. The accuracy should run better than 1% and the incremental accuracy better than ½% in the total environment experienced.

D. Payload Attitude Control

Control of the angular orientation, or attitude, of a space vehicle may be required for (1) guidance observations, (2) scientific measurements, (3) radio communication, (4) aiming solar cells at the sun, and (5) maneuvering. Attitude-control subsystems for use with comparatively high-thrust chemical rocket stages are referred to as autopilots and are not discussed here. The present discussion refers only to subsystems intended to operate during periods of low thrust and/or coasting flight (when no propulsive or maneuvering thrust is applied).

The function of an attitude-control system is to maintain prescribed relationships between reference directions in the controlled body and external reference directions. An attitude-control system has four major subdivisions, as shown in Fig. 31. The attitude-sensing devices may include radiation detectors, such as photocells, and inertial elements, such as gyroscopes. The latter type of device is limited by its inability to maintain a reference direction over a long period of time. The radiation detector can maintain a reference direction over an indefinite period, as can a sun-seeking device. Furthermore, accurate determination of direction is possible with optical devices, provided they are not hampered by the size of an extended source. However, the problem of identifying the radiation source to be used as a reference does exist.

The control computer relates the changes in attitude information to the appropriate corrective actions. The computer may contain both linear and nonlinear operations, the specific arrangement being determined by the mission of the vehicle.

The momentum-changing device adds or removes angular momentum from the hull as changes in its angular velocity are commanded by the control computer. Examples of such a device are sets of flywheels or of gas jets. The energy supply is necessary for the operation of the sensing equipment and control computer, as well as for producing the control torques that change the angular momentum of the hull. This energy might be supplied by solar cells or other electrical sources, supplemented by tanks of gas or monopropellant.

1. Methods of obtaining control torques. Three methods of obtaining control torques have been investigated: (1) gravitational field effects, (2) removal of angular momentum from the vehicle, and (3) transfer of angular momentum between the hull and attached devices. The latter two types of system will be referred to in the following material as removal and transfer systems, respectively.

Because the gravitational field of a celestial body is inversely proportional to the square of the radial distance from the body, the attractive forces on particles of equal mass at different radial distances are not the same. Hence, for a body with a nonuniform mass distribution, there exists a spring-like restoring torque which tends to keep the body aligned in a particular direction relative to the local vertical. For a body of revolution, this torque is proportional to twice the angle between the axis of revolution and the local vertical. The restoring torque is also inversely proportional to the cube of the radial distance from the body; hence, it is only useful at relatively small radial distances for such tasks as satellite attitude control. At the surface of the earth, for example, this torque is capable of causing angular accelerations of up to 1.5×10^{-6} rad/sec^2.

A distinction can be drawn between attitude-control systems intended only to maintain the angular attitude of the vehicle in a fixed direction with respect to a given reference and systems that are intended to vary the attitude of the vehicle in accordance with commands. Systems utilizing spin stabilization or differential gravity effects are of the former class, and are useful only in rather specialized cases. Command systems, at least about

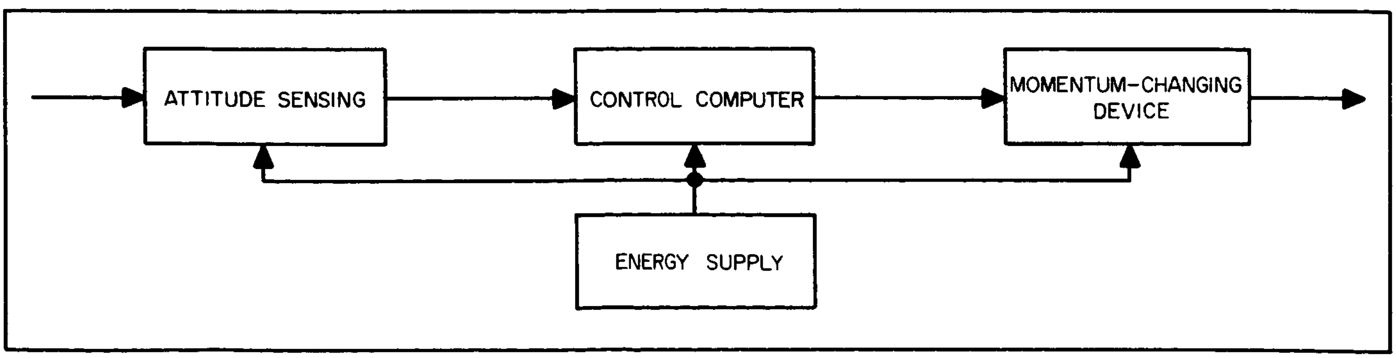


Fig. 31. Attitude-Control System

certain axes of the vehicle, will be required for most space missions. Some operations that utilize a command-type attitude-control system are listed in Table 2.

For a command system, a controllable energy source not dependent on the attitude of the vehicle is necessary. Such sources include a nuclear electric power supply; tanks of pressurized gas or propellant; and, in many instances, a solar–electric power supply. The required torques may be obtained from the expulsion of mass, or by accelerating or precessing flywheels. Of the several factors affecting the choice of energy supply, weight is one of the most important.

Studies of vehicles in the 300- to 1000-lb class have led to the following tentative conclusions regarding the choice of energy supply: The studies are based on the use of pressurized gas for momentum removal and of accelerated flywheels for momentum transfer. Torque levels of 0.2 ft/lb or less are sufficient for the attitude control of a 1000-lb vehicle during coasting flight if no rates in excess of 0.05 rad/sec about any axis are required. These levels are too low to justify the use of a hot-gas system. To date, no comparison has been made between accelerated and precessed flywheels.

In most cases, the attitude-control system will contain a gas-jet system. For voyages of more than 60 to 180 days during which the attitude is held constant or changes at rates less than 6 deg/hr, it is economical (weightwise) to use a momentum-transfer system for attitude maintenance, reserving the removal portion of the system to make more rapid maneuvers and to cancel the effects of initial conditions and disturbances. A transfer system alone is feasible from the weight standpoint only when the initial rates and rates built up as a result of external disturbances are extremely small. For the removal of an initial rate, a momentum-removal system is more economical (weightwise) than a transfer system because, in the latter type of system, conservation of angular momentum necessitates a permanent change in flywheel speed or orientation in accelerated and precessed systems, respectively. For the accelerated system, this results in added weight to take account of the speed bias, and for a precessed system, the effective torque axis is displaced.

For attitude maintenance, however, a transfer system is preferable because the gas-jet expulsion system is essentially an on–off device subject to limit cycles. Although one type of control computer (discussed in a succeeding paragraph) has been studied that is quite economical with respect to gas and electrical power, the momentum-transfer method of attitude maintenance becomes competitive for voyages of 60 days or more.

A preliminary study has shown that gaseous nitrogen stored at a temperature of 40°F and an initial pressure of 300 psia has many desirable properties for the momentum-removal portion of an attitude-control system for coasting flight.

2. *Low-energy digital control.* The weight of an attitude-control system increases as the average magnitude of the angular rate of the vehicle becomes larger because greater changes in the angular momentum of the hull must be effected. For this reason, it is desirable to limit angular rates to low values whenever possible. The necessity of limiting both the power level and energy consumption and the difficulty of measuring small rates add to the problem of attitude control during the major portion of the voyage, when the primary task of the system is to keep the vehicle pointing at a specified reference.

Table 2. Attitude-Control Accuracy Requirements

Operation	Position Accuracy deg	Rate Accuracy deg/sec
Mid-course maneuver	±0.25 to 2	
Satellite measurements		
Satellite observatory	±0.5	
Meteorological	±1.0	
Deep-space measurements		
Photographs of planets	±0.1 to 1.0	±0.001 to 10.0
Navigation sightings[a]	0.005 to 1.0	
Communication		
20-db-gain parabolic antenna	±8	
36-db-gain parabolic antenna	±1	
Solar power	±10 to 20	

[a]Accuracy depends on extent to which vehicle attitude reference and control are an integral part of navigation system.

A method of attitude maintenance that diminishes the power level and weight of the system has been studied. The system requires neither rate information nor 3-bit digital position information to maintain a slow, stable limit cycle or to follow slow changes in the attitude-reference direction. The operation of this digital attitude-maintenance system is described below, with the aid of Figs. 32 through 35. Figure 32 is a phase-plane diagram of angular position of the vehicle ϕ_v vs angular rate ω_v about a single axis, with the origin at zero rate and zero displacement from attitude-reference direction.

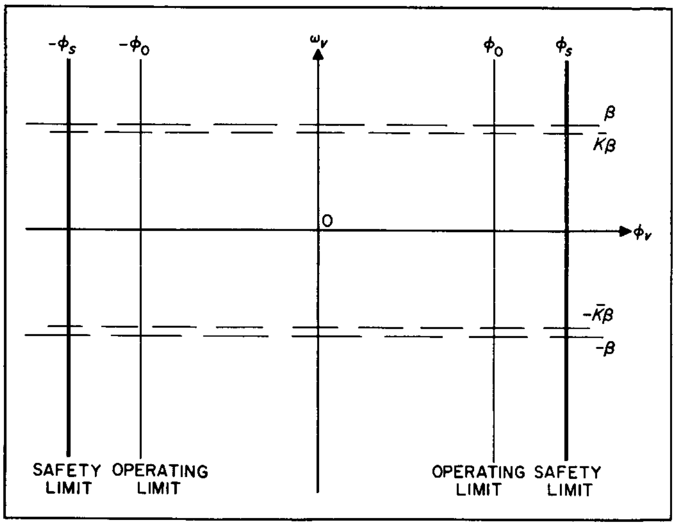


Fig. 32. Phase-Plane Diagram Describing Digital Attitude-Maintenance Operation

The operation of the system may be described as a slow limit cycle between operating limits $\pm\phi_0$ (Fig. 32) of angular displacement, with an occasional excursion to a larger safety limit $\pm\,\phi_s$ as a result of disturbances, system unbalance, or change in the reference direction. The period of the limit cycle is kept long by making calibrated velocity changes of magnitude β at the operating limits, or of magnitude $\bar{K}\beta$ at the safety limits, where $0 < \bar{K} < 1$. These changes are only large enough to provide a reversal of motion and a small drift velocity in the opposite direction. This method of operation requires that the initial conditions be reduced to an angular displacement from the reference smaller than the safety limit, and a rate error less than β in magnitude.

The details of operation when no disturbances are present are described with the help of the phase diagram in Fig. 33. Assume that the point (the space probe SP) representing the state of the vehicle is initially at A. Then, if no disturbing torques are present, the velocity remains constant and the SP moves to B. Here, a change in velocity of $-\beta$ is introduced, moving the SP to C. Because the velocity at A is less than β, the velocity at C is of the opposite sign. It again remains constant, and the SP moves to D. Here, a change in velocity of β is forced, moving the SP to E. The velocity is now the same as it was originally, and the SP moves through A, the cycle repeating itself.

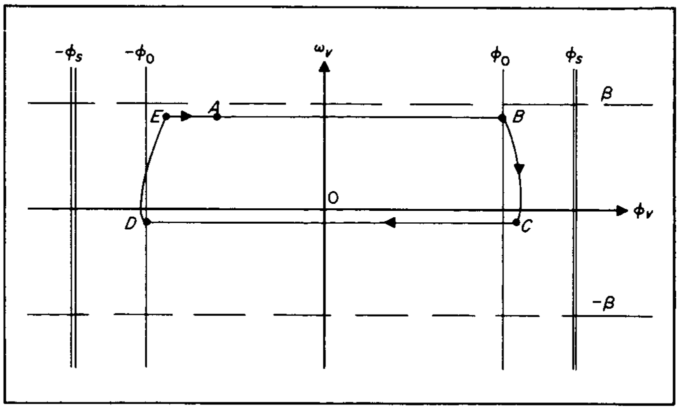


Fig. 33. Attitude-Maintenance Operation Without Disturbances

If disturbing torques are present, the safety limits may be reached. The operation is illustrated in Fig. 34. Assume that the SP is initially at point E. If a constant disturbing torque is present, the SP will move to F. Here, a change in velocity of $-\beta$ is forced, moving the SP to G. Because the initial velocity was greater than β in magnitude, no reversal of sign in velocity occurs, and the SP moves past the operating limit toward H at the safety limit. Provided

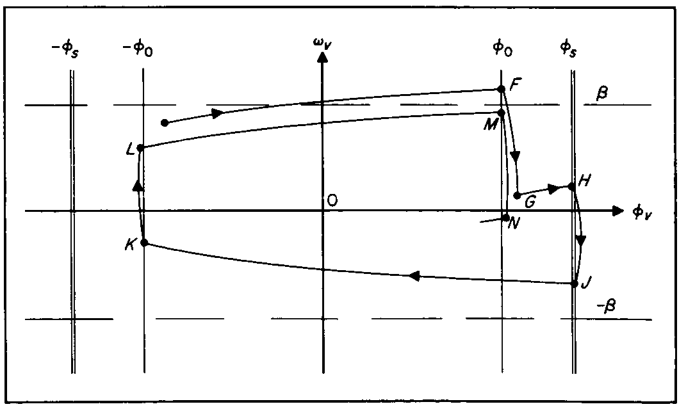


Fig. 34. Attitude-Maintenance Operation With Disturbances

that a certain relationship between ϕ_0, ϕ_s, and the disturbing torque is satisfied, the velocity at H is less than $K\beta$, where $0 < K < 1$. At the safety limit, a velocity change of $-K\beta$ is forced, moving the SP to J, where $K < \overline{K} < 1$. Hence, the velocity changes sign, and the SP moves toward $-\phi_0$. Again, provided that a certain relationship is satisfied, the velocity remains negative, so that the SP moves to K where a change of β is forced, and the SP moves to L For sufficiently small disturbances, the magnitude of the velocity does not increase beyond β before ϕ is reached, and at M a velocity change of $-\beta$ moves the SP to N. The same events which occurred after the SP reached J are then repeated, but it can be seen that the entire phase-plane path has been displaced in the positive velocity direction. Eventually, a condition similar to the process from E to F occurs, and the velocity at ϕ_0 is greater than β. Hence, the safety-limit forcing has the effect of resetting the normal operating cycle.

If the disturbing torques are greater than a certain magnitude, it is possible to establish a limit cycle operating from $-\beta/2$ to $\beta/2$, wherein the SP never reaches $-\phi_0$. Other more complicated behavior is also possible, in which all of the forced velocity changes are of the same sign. The saturating effects of this behavior on a momentum-transfer type of system must be considered if such a system is contemplated, and provision made for the removal of such stored angular momentum with a mass-expulsion device.

Figure 35 is useful in discussing a possible mechanization for the attitude-maintenance equipment. It is assumed that an optical sensing device is used, and that the celestial body toward which the body must be pointed is illuminated and sufficiently distant so that its angular diameter is smaller than the distance between either ϕ_0 or $\phi_s - \phi_0$, whichever is smaller.

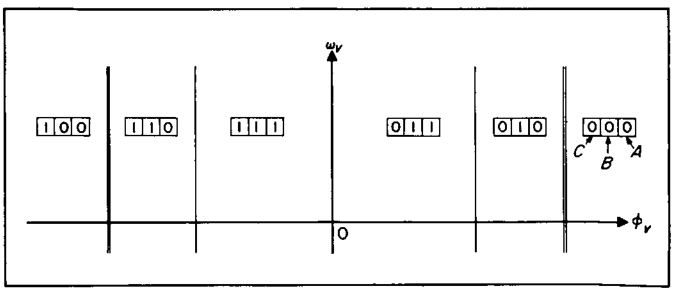


Fig. 35. A Possible Mechanization for Attitude-Maintenance Equipment

The discussion in the preceding paragraphs indicates that a corrective action must be taken whenever the magnitude of the angular displacement becomes greater than ϕ_0 or ϕ_s. Also, negative displacements require different action than positive displacements. Hence, the attitude sensor need only be a device that changes state when any of the following events occurs: (1) displacement is larger in magnitude than ϕ_0, (2) displacement is larger in magnitude than ϕ_s, and (3) displacement is positive.

The various possible states may be represented by a 3-bit word, as illustrated in Fig. 35, where conditions C, B, and A are represented by the left, middle, and right digits, each of which is zero when the condition listed above occurs. Figure 36 illustrates a possible sensing device. The rays of light from the celestial body are focused into a narrow elliptical spot, with the major axis parallel to the axis about which control is being maintained. Strips of photosensitive material are applied to the plate in the manner shown, and each strip is used to drive a relay or other switching device. Figure 37 shows

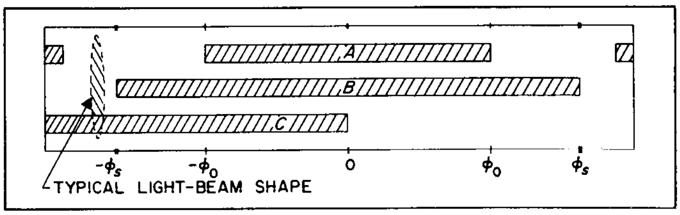


Fig. 36. A Possible Optical Sensing Device for Attitude-Maintenance System

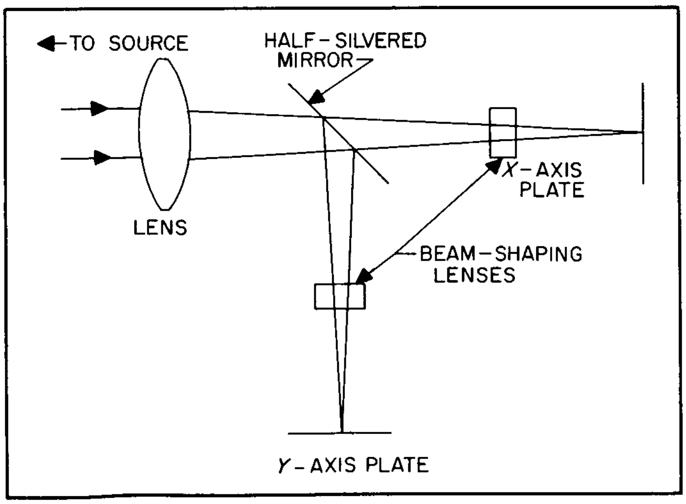


Fig. 37. A Possible Optical Arrangement for Pointing the Vehicle at a Celestial Body

a possible optical arrangement for pointing the vehicle at a celestial body, a task requiring two-axis control.

For proper system operation, once the displacement has increased in magnitude beyond ϕ_0 or ϕ_s, the velocity change commanded by such a state change should take place only once during the interval between the occurrence of the state change and the time when the displacement is again less than ϕ_0 in magnitude. This may be accomplished by using additional relays and contacts on relay *A* to form lockout circuits.

E. Space Navigation and Terminal Guidance

For future long-range space missions, the injection conditions cannot be controlled to sufficient accuracy. Subsequent correction of the coasting trajectories, referred to as mid-course or terminal guidance, will be necessary. Terminal guidance is also taken to include braking maneuvers to achieve, for example, a soft landing on the moon.

The operation of most systems of trajectory correction is based upon linear perturbation theory, i.e., the assumption that an actual trajectory differs only slightly from some previously computed standard trajectory. From that theory it can be shown that, to correct an actual trajectory to hit a moving point in space at a given time, the three components of the correcting velocity applied to the vehicle are calculable in terms of linear combinations of the perturbations (from the standard trajectory) in position and velocity. However, the desired end condition is usually to hit the moving point irrespective of time. In that case, there is a degree of freedom in the correcting velocity vector; e.g., the direction of the vector can be chosen to minimize its magnitude. Apart from economizing on rocket propellant, this also makes any miss (at the destination) a second-order function of angular errors in orienting the thrust vector.

It is implied that the actual trajectory differs from the standard only because of injection errors; in other words there is no analog of mid-course disturbances, such as wind, as in the case of short-range surface-to-surface missiles.

1. Guidance systems. In considering schemes of guidance, closed-loop systems (such as those used for the guidance of homing missiles) have been ruled out. Continuous correction of the trajectory would be too expensive in terms of rocket propellant for space missions. The three systems seriously considered would correct the trajectory by one or two discrete-impulse, open-loop corrections. The schemes are characterized by the method of measurement: (1) radio measurements from the earth, (2) optical sightings from the vehicle, and (3) angular-rate measurements of the target body relative to the vehicle. The power requirements for active radio-echo guidance are prohibitive, and the flight times are too long for all-inertial guidance. Methods (1) and (2) would probably constitute mid-course guidance; in order that the angular rates be measurable, method (3) would, in practice, be used for terminal guidance.

It is desirable that any trajectory be corrected as early in flight as possible; the effect of the weight of rocket propellant is thus minimized, but, unfortunately, the early measurements tend to be more exacting. Table 3 shows typical data for the correction of various trajectories.

For a ground-based radio tracking and command system, the critical measurements would be the azimuth

Table 3. Typical Propulsion Data for Trajectory Correction

Mission	Typical Target Spread Without Correction miles	Distance from Target when Correction Applied miles	Correcting Velocity ft/sec	Rocket Propellant and Chamber % payload ($I_{sp} = 265$)
Moon	±1,000	10×10^3	420	7.2
		20×10^3	210	3.6
		12×10^4	32	0.55
Mars	±400,000	30×10^6	270	4.5
Venus	±250,000	30×10^6	83	1.4

and elevation angles. To keep the final dispersion at the moon down to $\pm$ 25 to 50 miles would require angular accuracies of $\pm$ 0.1 mil.

To determine position by optical measurements from the vehicle (space navigation), sightings would be required on the moon, the earth, and two stars, one of which might be the sun. For an interplanetary mission, two planets would be sighted relative to their star background. For a final dispersion at the moon of $\pm$ 25 to 50 miles, mid-course sightings should be accurate to about 0.07 mil; a dispersion at Mars of 20,000 to 40,000 miles would correspond to sighting errors of about 0.1 mil at 30 million miles from Mars. Of course, subsequent corrections may be made much nearer Mars.

Although space navigation by optical sightings seems promising for some interplanetary missions, it does not appear feasible for lunar missions since the sightings would be on bodies of non-uniform brightness (or only partially illuminated) subtending large angles. For example, at 120,000 miles the earth subtends 67 mils; bearing in mind that it may be only partially illuminated, it would hardly be practicable to locate the center to 0.1%, the required accuracy. For this reason, terminal guidance using angular-rate measurements has been studied.

Such a system would have an optical sensor fixed in the vehicle, and the vehicle, or perhaps just the guidance package, would be rotated to keep the optical axis pointing at the center of the target. The correcting-velocity vector is calculable in terms of the range (from the angular diameter) and from the measured angular rates of the vehicle.

2. ***Post-injection radio guidance.*** A system of early mid-course guidance based on radio measurements and command from the ground is attractive because vehicle equipment would be simple with most of the complications on the ground. However, the key factor in assessing the feasibility of such a system is whether the available accuracies of measurement meet the required accuracies.

In order to make a trajectory correction the six coordinates of position and velocity must be known, but they need not be measured as such. On a calculable ballistic trajectory with no unpredictable disturbances it is merely necessary to compute the appropriate number of boundary conditions for the second-order differential equation which determines the motion of the vehicle in space. Thus, in considering the radio guidance of a vehicle some three days after injection into an interplanetary orbit, it would be feasible to measure only range rate, elevation and azimuth. The measurement of range may require a wide-bandwidth radio link (or a complicated procedure for integrating range rate) and the angular rates would be inconveniently small for measurement, e.g., 0.05 microrad/sec. On the other hand, range rate is conveniently measured by radio-doppler techniques with a transponder in the vehicle. Some numerical analysis was therefore carried out to determine how accurately it would be necessary to measure range rate, elevation, and azimuth for guidance with only these quantities.

Taking the example of a 165-day Mars trajectory, with a trajectory correction 1 million miles from the earth, the miss at Mars due to measurement errors was computed:

Range rate	4,000 miles/ft/sec
Elevation angle	20,000 miles/millirad
Azimuth angle	30,000 miles/millirad

In measuring range rate by a doppler system, the limiting accuracy would probably be a steady drift of the reference frequency source on the ground during the 10-sec interval that the signal takes to cover twice the 1-million-mile distance. Even so, a frequency drift of 1 in 10^5 in 10 sec—achievable in practice—would be sufficient to make insignificant the effect of range-rate errors.

The accuracy of the JPL Goldstone 85-ft-diameter antenna is currently reported to be about $\pm$0.3 millirad; ultimately it will be about $\pm$0.1 millirad by taking advantage of the world tracking net.

The real-time tracking procedure, described in Section III-H of this Report, is capable of fitting the tracking data to give a smoothed estimate of either angle with a standard deviation of less than 0.1 times the uncertainty in any single-position measure. However, uncertainties in the systematic errors (boresight errors) may make this deviation somewhat greater.

Using an optimistic figure of $\pm$0.1 millirad for the smoothed angular errors, the miss at Mars would be 4000 miles. This figure is considerably less than the error ($\sim$50,000 miles) due to the uncertainty in the astronomical unit, to which earth-bound guidance systems are susceptible. The accuracy corresponding to 0.1 millirad in the angles could not, therefore, be fully realized until the astronomical unit were known more accurately, presumably from a previous space-probe experiment or from radio reflections off a planet as recently reported by Lincoln Laboratories.[1]

As regards the rocket which will make the correction, it has been shown that it could conveniently be positioned to lie with its thrust axis in a plane normal to the vehicle-to-earth line, that plane being not too different from the "critical plane" which minimizes the magnitude of the necessary correction. By this means, one attitude reference—a directional communication antenna—could keep the roll-axis constantly pointing at the earth. A sun- or moon-finder would probably constitute the other attitude reference. Also, there would have to be the facility for rotating this sun-finder according to a radio command in order to point the rocket, the latter being mounted to fire along the pitch or yaw axis.

Although it would be simpler to employ a variable-impulse rocket, fixed-impulse solid rockets cannot be overlooked. It has not yet been ascertained whether it would be feasible to delay the firing time and use a fixed-impulse rocket. A third alternative, varying the firing direction, would complicate the system of attitude references.

With radio tracking accurate to 0.1 millirad, such a system would probably be not limited by the accuracy of angular measurements, but rather by the accuracy of orienting the rocket.

Most of the weight of such a system would be in the rocket motor; this weight would depend on the accuracy of the injection-guidance system, i.e., the miss that would result if there were no correction to the trajectory. For example, if the uncorrected miss at Mars were 250,000 miles, the combined weight of rocket propellant and chamber would account for 2.6% of the payload.

3. Self-contained guidance systems. Two systems of self-contained navigation have been studied: (1) one which takes angular sightings on two planets, using stellar reference directions, and (2) one which sights only the destination planet relative to its star background, and determines range from the angular diameter of that planet.

Angular sightings on planets and stars constitute measurements to determine the position of a space vehicle. Such measurements to determine position might be supplemented by velocity measurements employing optical passive doppler techniques, but it seems extremely doubtful at this time whether such measurements, if practicable, would be sufficiently accurate. In any case, as explained in the previous paragraphs, it is not necessary to determine all six coordinates of position and velocity as such; two independent determinations of position suffice to determine the correcting velocity vector. Assuming that actual trajectories differ only slightly from the standard, linear perturbation theory may be applied, most of the theory then being linear matrix algebra.

Referring to Case 1 above, it is assumed that one axis of the vehicle is kept pointing at the sun (or earth) to a moderate accuracy. From the sightings on two planets relative to their star background, two vectors may then be drawn from each planet to intersect at the space vehicle. A preliminary inspection of star maps of the constellations of the Zodiac indicates that it should be possible to use stars at least as bright as the third or fourth magnitude which lie not more than 5 deg from the planet in question. Two angular coordinates (ψ, α) could then be measured which would define the angular position of the planet relative to the reference star. The angle ψ could be obtained by traversing a telescope axis in the plane containing the lines from the vehicle to the sun and planet (Fig. 38); the other angle α would be from a traverse in a perpendicular plane containing the line joining the vehicle and planet. It would be advantageous to traverse the telescope axis rapidly so that any slow drift of the vehicle would not cause a significant error in the measurement; the demands on attitude control of the vehicle would not then be so exacting. The angles ψ and α are shown in Fig. 38 for two planets P_1 and P_2.

Some results of the error analyses on a 140-day Mars trajectory are tabulated below. The angular sighting errors were assumed to have the same magnitude but are uncorrelated. Thus, in a typical situation where the first corrections were applied 14 million miles from Mars, 10,000-mile accuracy at that destination would correspond to optical sightings accurate to about 0.1 millirad. The relative positions of the planets sighted at the times tabulated are shown in Fig. 39.

Time from injection days	Distance from Mars miles $\times 10^{-6}$	Miss miles/millirad of angular error
50	51	266,000
110	14.4	84,500
130	5	51,700

[1]Price, R., et al, "Radar Echoes from Venus," *Science*, 129:751, 1959.

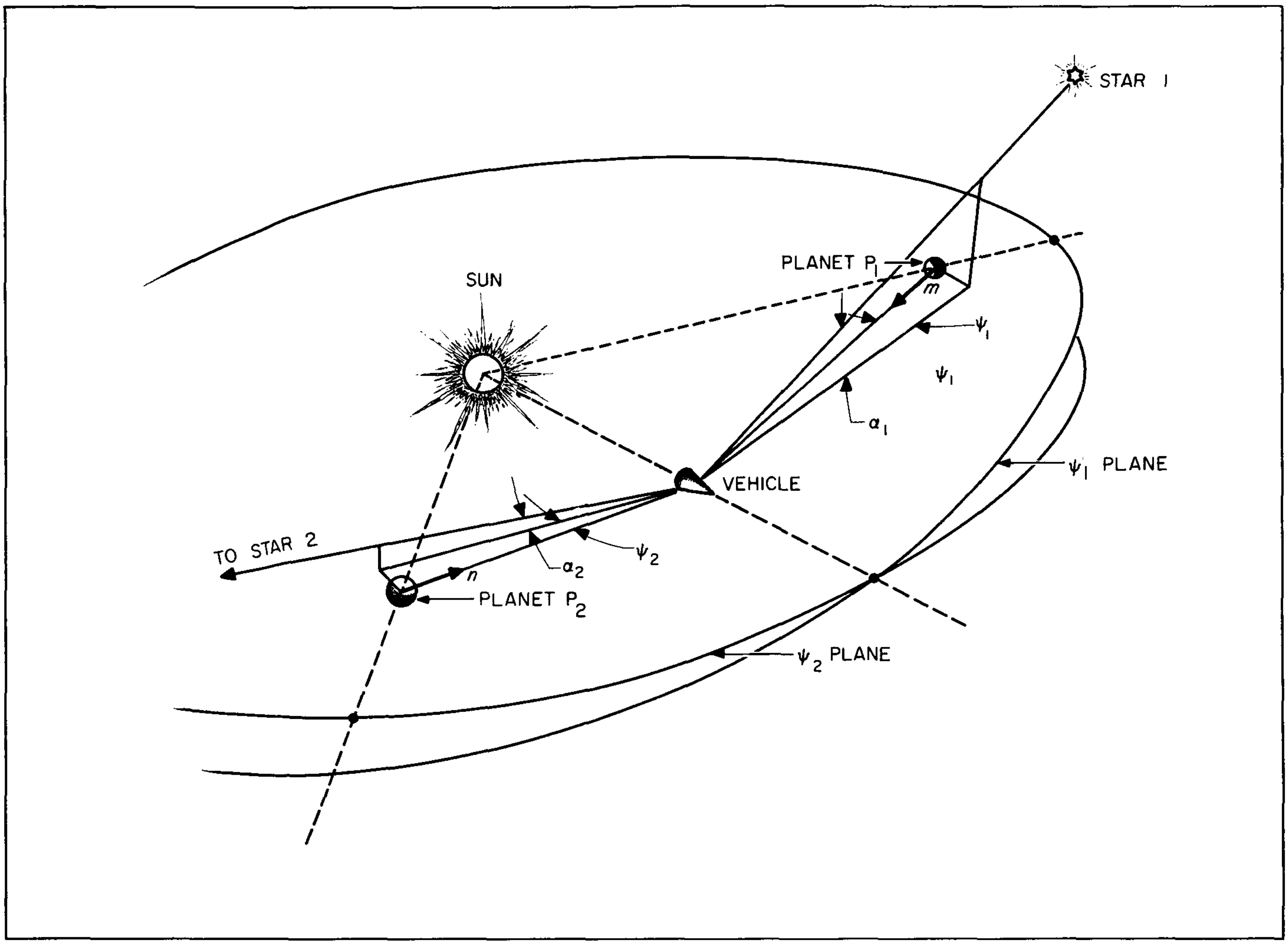


Fig. 38. Measurement of Angular Coordinates (ψ, α) to Define Angular Position of Planet Relative to Reference Star

Case 2 refers to a method which sights on the destination planet as in Case 1 but the only other measurement would be on the angular diameter of the planet, in order to give range. Some pertinent results for this case are tabulated below. It then becomes necessary to know how accurately it might be possible to measure the angular diameter.

Time from injection days	Distance from Mars miles $\times 10^{-6}$	Miss miles per % range error	Miss miles/millirad of angular error
110	14.4	37,600	24,400
130	5	6,400	7,400

The fundamental limiting accuracy of the optical system would be fuzziness of the image due to diffraction by the optical aperture, an effect which depends on the wavelength of light detected. Unfortunately, Venus will present a crescent phase to an approaching vehicle; this suggests that it will be necessary to use infrared in order to see a circular disc. Calculations comparing the radiation from the light and dark sides of Venus and Mars indicate that the shortest possible wavelength will be 4 to 5 microns. Corresponding to this wavelength the percentage errors in the angular diameter are tabulated below. The measurement of range will be sufficiently accurate only for corrections within 1 or 2 million miles of the destination planet.

Planet	Distance miles $\times 10^{-6}$	Angular-diameter error %
Mars	5	5.3
Mars	1	1.1
Venus	5	2.9
Venus	1	0.6

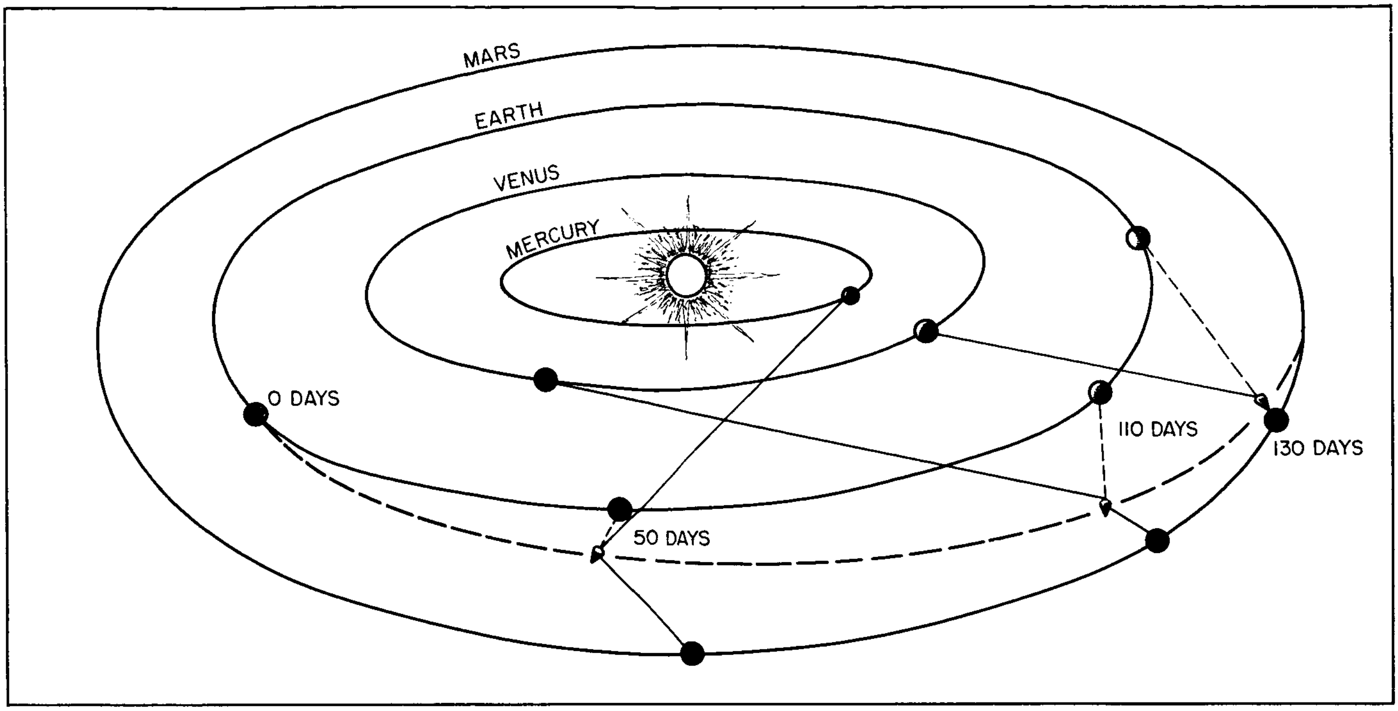


Fig. 39. Relative Positions of the Planets

For interplanetary navigation there seems therefore to be two possible lines of development, probably only the first being applicable to missions to the more distant planets:

(a) Use of the true celestial navigator (Case 1), which calls for sightings accurate to 0.1 millirad, to guide the vehicle to within 5,000 to 10,000 miles of Mars or Venus. An additional terminal-guidance system would be necessary.

(b) Use of post-injection radio guidance to ensure approaching within 10,000 miles of Mars or Venus, followed by "approach guidance" (Case 2) requiring sightings no more accurate than 1 millirad and range accurate to 1 or 2%. For lunar navigation, the post-injection radio guidance would suffice.

To achieve a landing in some designated area of the planet surface, or to establish a reasonably well-controlled orbit around a planet, the guidance methods so far described would not be adequate. Some additional "terminal" guidance maneuver would be required. Such a terminal-guidance method has been analyzed, with the moon as a representative target.

For this particular target, the moon, earth-based radio guidance is competitive with the capabilities estimated for the vehicle-borne terminal-guidance device. However, this situation does not apply in the case of planetary targets. The system described in the following paragraphs is equally applicable to planetary missions.

4. Terminal guidance to the moon. In guiding for a given closest-approach distance to the moon, the critical measurements would be the angular rates of the vehicle, since one axis of the vehicle would be kept pointing at the center of the moon. To make these rates sufficiently great to measure with rate-gyros, the correction would be applied 10,000 to 15,000 miles from the center of the moon; i.e., well within the lunar sphere of influence (radius 40,000 miles). The guidance equations then assume a simple form. The correcting thrust vector is in the trajectory plane and, with little loss of efficiency, can be at right angles to the lunar radius vector. The rate of turn $\dot{\alpha}$ and length r of that radius vector are then all that is required to compute the magnitude of the correcting velocity V_c,

$$V_c = \frac{-1}{r_{fire}} (r^2 \delta \dot{\alpha})\, r_{measure}$$

To mechanize this scheme (Fig. 40), an optical sensor fixed in the vehicle and pointed along the roll axis would be required, and the correcting rocket would be mounted

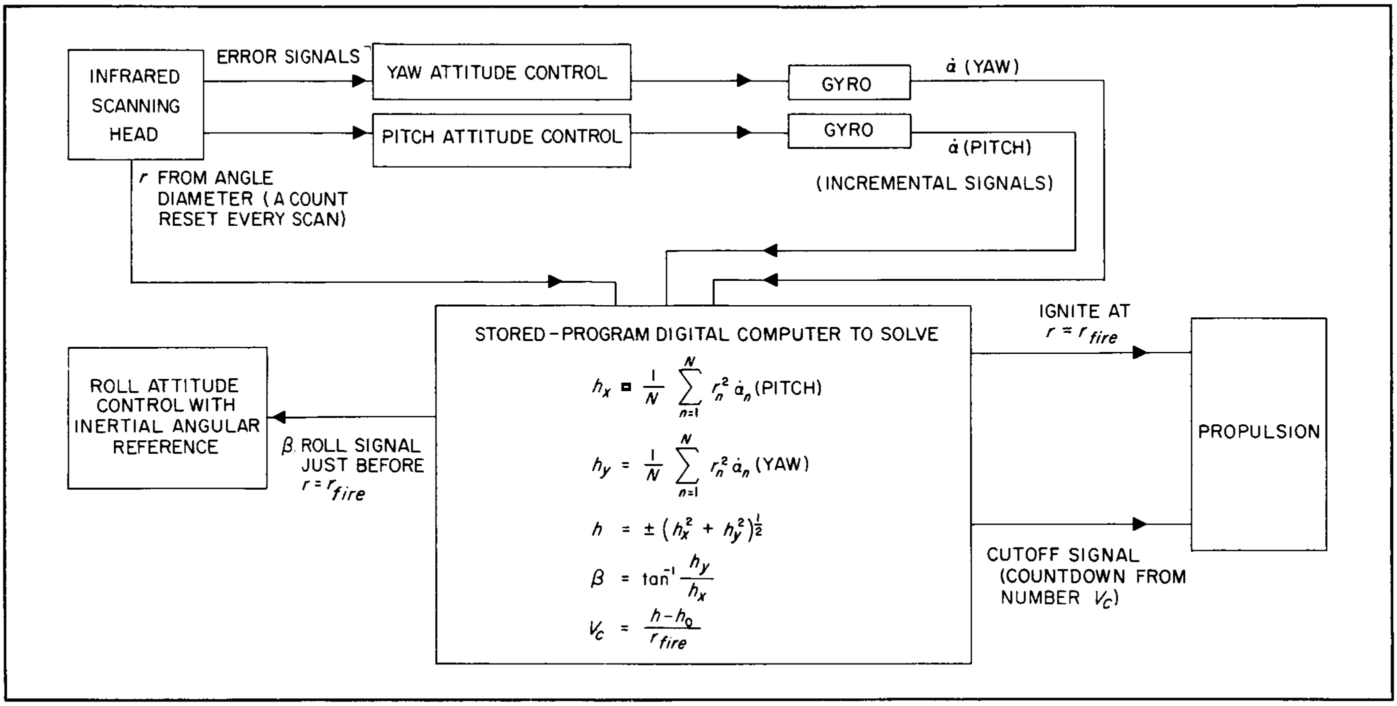


Fig. 40. Guidance for Closest Approach to the Moon

to fire along the pitch or yaw axis. The sensor would give the angular diameter of the moon and error signals by which the attitude control system would keep the roll axis pointing at the center of the moon. Over a measurement period (e.g., ½ hour), the vehicle would be rotated to keep the roll axis pointing at the moon; this vehicle rotation about pitch and yaw axis would be measured by rate-gyros. The vehicle would be stabilized in roll by sighting on the sun.

If the vehicle were rolled to keep the pitch axis perpendicular to the direction of the sun (by a suitable sun-finder), then the measured pitch and yaw rates $\dot{\alpha}_p$ and $\dot{\alpha}_y$ would be related to rates of the angles (θ, ϕ) of a spherical polar coordinate system, centered at the moon with $\dot{\phi}$ normal to the ecliptic plane. During the measurement phase, θ would be nearly 90 deg; however, trajectories in which the sun would appear in approximately the same direction as the moon (or opposite to it) when viewed from the vehicle would have to be excluded. Two separate sets of measurements would theoretically be taken to give the perturbations: ($\delta R'$, $\delta\dot{\theta}'$, $\delta\dot{\phi}'$) at about 20,000 miles and (δR, $\delta\dot{\theta}$, $\delta\dot{\phi}$) at 10,000 miles from the moon. To make a correction at 10,000 miles, ($\delta\dot{R}$, $\delta\theta$, $\delta\phi$) are also required. Assuming that first-order perturbation theory applies from earth injection to the measurement points, they can be expressed as linear combinations of (δR, $\delta\dot{\theta}$, $\delta\dot{\phi}$, $\delta R'$, $\delta\dot{\theta}'$, $\delta\dot{\phi}'$), the coefficients being a function of the standard trajectory. The equations defining the pitch and yaw components of the correcting velocity can similarly be expressed as linear combinations of perturbations in the measured pitch and yaw angular rates at the two measurement points. Thus for the pitch and yaw components of the correcting velocity:

$$V_p = K_1 \delta\dot{\alpha}_p + K_2 \delta\dot{\alpha}_y + K_3\, \delta\dot{\alpha}_p' + K_4\, \delta\dot{\alpha}_y'$$
$$V_y = C_1 \delta\dot{\alpha}_p + C_2 \delta\dot{\alpha}_y + C_3\, \delta\dot{\alpha}_p' + C_4\, \delta\dot{\alpha}_y'$$

where subscript p stands for pitch and y for yaw, K and C are coefficients determined from the standard trajectory, and the primes refer to the first rate measurements. Since measurements could be taken at $\delta R = 0$ and $\delta\dot{R} = 0$, δR and $\delta\dot{R}$ have been dropped. However, when taken well within the lunar sphere of influence, two separate sets of rate measurements become superfluous, and the equations to be mechanized are then approximately:

$$V_p = K_1^* \dot{\alpha}_p + K_2^* \dot{\alpha}_y$$
$$V_y = C_1^* \dot{\alpha}_p + C_2^* \dot{\alpha}_y$$

Correcting velocity

$$V_c = (V_p^2 + V_y^2)^{\frac{1}{2}}$$

Roll angle

$$\beta = \tan^{-1}\frac{V_p}{V_y}$$

A floating gyro would be suitable for the rate measurements. (The same three gyros might have been used for injection guidance.) The drift rate for trimmed MIG gyros is expected to be the order of 0.03 deg/hr; such an error would cause a ± 18-mile error in the closest-approach distance. Again the attitude-control system in pitch and yaw should have a rate accuracy of about ± 0.05 deg/hr.

The optical head should be able to sense the center of the moon whatever the state of illumination, even for a crescent moon. To accomplish this, the use of the far infrared spectrum seems most promising, in order that the whole moon can be seen. The temperature of the bright side is about 100°C and that of the dark side -100 to -150°C. These temperatures dictate that the wavelength region of about 10 to 40 microns should be used. The brightness ratio from the hot to the cold side would then be about 60:1.

Photoconductive detectors such as used in the detection of aircraft are inapplicable for the far infrared spectrum; instead, a thermal detector should be used. Because no suitable image tubes are likely to be available in the next few years, a mechanically scanned single-thermistor detector would be employed, the latter being commercially available. It is anticipated that the angular diameter of $\pm 1\%$ and the center to $\pm 1\%$ of the angular diameter will be achieved. Such figures are acceptable.

Automatic computation of the equations has been considered by analog computer, digital differential analyzer (DDA), and special-purpose stored-program digital computer. From the standpoint of weight and power, the latter is the most promising approach. Magnetic storage is indicated because a continuous drain of power is not associated with the memory. Although access is simpler for a drum, core-matrix storage would probably be lighter to store the relatively small number of words required in this computer. In such a special-purpose computer the program is in fixed storage; according to work at the MIT Instrumentation Laboratory, the number of words can then correspond to the number of cores and the number of bits per word to the number of wires threading each core. The erasable storage cannot be constructed in this fashion, but even so it has been estimated that no more than 200 cores would be required for the whole computer (including working storage, the arithmetic, etc.).

a. Mechanization for a lunar satellite. After firing the correcting rocket at 10,000 miles from the moon, the vehicle would make the closest approach to the moon 2 hours later. To establish the vehicle in a permanent (approximately circular) orbit 700 miles above the lunar surface would require a velocity reduction of 3400 ft/sec. This could conveniently be achieved by a fixed-impulse solid-propellent rocket, accounting for about 38% of the payload, as indicated in the following tabulation:

Rocket	Weight payload %
Correcting rocket (420 ft/sec)	
Propellant ($I_{sp} = 265$ sec)	5.0
Inert weight	2.2
Retro-rocket (3400 ft/sec)	
Propellant ($I_{sp} = 265$ sec)	33
Inert weight	5

Before firing the retro-rocket, the vehicle would be rotated through a given angle using the inertial references established by the earlier optical sighting. Firing would be at a given time after the correction point.

Distortion of the attempted circular orbit due to various errors at lunar injection has been studied. It is estimated that the actual orbit of the lunar satellite could be confined between two concentric spheres, 530 and 860 miles above the surface of the moon.

b. Mechanization for soft landing on the moon. Assuming a slow coasting trajectory (70 to 80 hours), the impact speed would be about 8500 ft/sec; to cancel this speed using a liquid propellant with a specific impulse of 290 sec will require 60% of the payload as propellant and about 11% for the motor and tanks. There are, however, two major problems associated with the soft-landing mission: (1) the velocity must be brought to zero near the surface of the moon; e.g., not more than 100 ft above the moon, and (2) the retro-rocket axis must be aligned accurately with the final velocity vector. With reference to the former problem, to control ignition of the retro-rocket the altitude would be required to an accuracy of about ± 40 ft; controlled shutoff also would be necessary. However, due to variations in burning time (thrust level), zero velocity might occur as high as 4000 ft. The vehicle would then fall and hit the moon at 200 ft/sec. It follows that, for true soft impact (speed less than 50 ft/sec), some vernier control of braking is necessary. One such scheme is presented in Fig. 41.

The main motor would be ignited at a given height, and vernier control (to simultaneously obtain a height and speed of zero) would be realized by changing to a low-thrust phase at variable height of a few hundred feet. If h_0 and V_0 are the height and speed at the point of changing to the low-thrust phase,

$$\tfrac{1}{2}V_0^2 = (b - g)h_0$$

where g is the gravitational acceleration at the surface of the moon and b the braking retardation. The mechanization for the thrust-reduction signal is therefore based on the differential of the equation:

$$V_0\,\delta V_0 = (b - g)\,\delta h_0$$

Final cutoff would occur near zero height.

If continuous variation of thrust is available it might be advantageous to control the retardation $(b - g)$ according to

$$(b - g) = K\frac{V^2}{h}$$

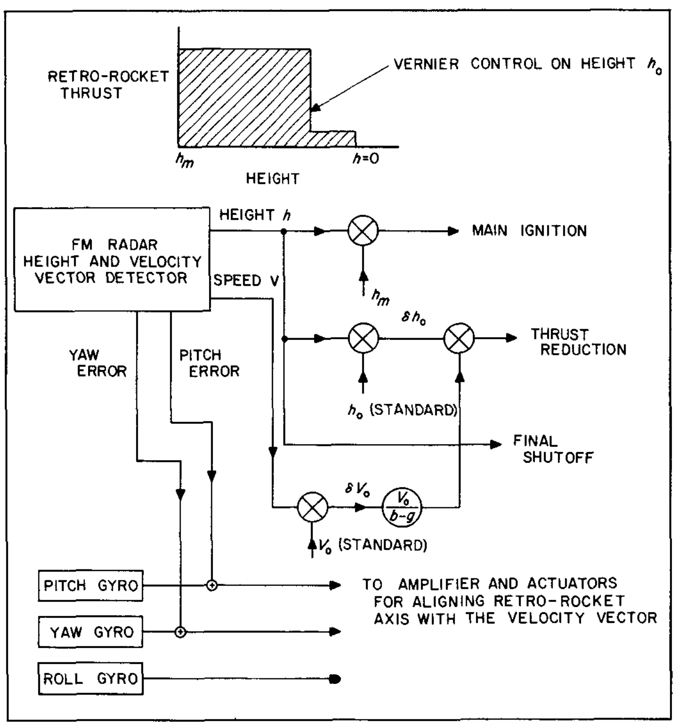


Fig. 41. Vernier Control of Braking for Soft Impact

By making $K >> \frac{1}{2}$, the final retardation would be gradual and the performance of the system less susceptible to measurement errors in V and h.

F. Communication Problems and Capabilities

Minor modifications in existing Microlock communication facilities would make it possible to receive tracking signals and scientific payload information from a distance of about two million miles. The moon is the only celestial object within this range. The next logical step from an exploratory point of view is the development of communication techniques for a range in excess of a hundred million miles with a power supply weighing approximately 100 lb. The areas in which improvements are called for are power sources, vehicle antennas, earth-based amplifiers, and microminiaturization techniques. These improvements would not be such formidable problems if it were not assumed that communications systems had to be much more reliable than other component systems of the vehicle.

The weight required to transmit vehicle and scientific information is proportional to the expression

$$2\pi\int\frac{KTMLR^2}{EDA_tA_r}\,dt$$

where:

$K \equiv$ Boltzmann's constant

$T \equiv$ effective temperature of the receiver detector

$M \equiv$ sampling rate of the experiment

$L \equiv$ effective signal-to-noise ratio desired from the receiver

$R \equiv$ distance of information transmission

$E \equiv$ effective efficiency of transmission system

$D \equiv$ storage capacity of batteries

$A_t, A_r \equiv$ effective areas of the transmitting and receiving antennas respectively

and the integral is taken over the time of transmission. The product $KTMLR^2$ is directly proportional to the power output of the system.

This formula demonstrates that a space program which envisions making increasingly refined experiments at ever greater distances must either pay the price of heavy power supplies or must undertake significant increases in efficiency, storage capacity, and effective antenna area, as well as a significant decrease in the effective receiver temperature. These last two factors will now be given more detailed attention.

At present the effective noise temperature T of the receiving antenna at Goldstone is estimated at 65°K. The temperature breakdown is:

Source of Contribution:	*Contribution, °K*
Sidelobes from aperture pattern	22
Feed spillover	8
Transmission through dish surface	6
Ohmic losses (0.5 db assumed)	29
Total antenna noise temperature	65

Additional assumptions are that the antenna sidelobes have an average level of 8 db below isotropic or 50 db below the antenna maximum, and that the surrounding ground temperature was 290°K. The sky temperature is neglected.

The ohmic-loss noise temperature may possibly be reduced to a few degrees Kelvin by using a suitable transmission line and antenna-feed design. The sidelobe contribution might be reduced by surrounding the antenna with a metallized fence and metallized ground surface. These two features in themselves would decrease the weight required above by a factor of 2 or 3.

As indicated by the weight expression, large-area antennas are an important part of the vehicle communication system. Of course, the antenna has some weight due to its own structure, but this weight need not be very great for a vehicle coasting in free fall through space. The strength of the structure must be great enough only to operate as it goes through the opening maneuver (it is assumed that these antennas will not be opened until the vehicle has actually reached free fall), and to maintain its shape against any restoring forces of the material itself.

To a certain extent, a large effective antenna area can be obtained with a small physical antenna area through the use of large antenna gain G. Furthermore, if the antenna gain is fixed, then the effective antenna area can be increased by decreasing the transmission wavelength λ according to the relation $A > G/4\pi\gamma^2$. Therefore the eventual use of higher frequencies is anticipated.

The gain of an antenna of given size is limited by its surface roughness and its directional control. An imperfect surface on a parabolic reflector creates phase errors in the signal being reflected from its surface. The resulting interference causes a decrease in effective signal strength. In general, the mechanical surface errors tend to increase in proportion to the antenna size. Thus, larger antennas present more difficult problems in structural design.

The antenna gain of a parabolic reflecting antenna is obtained by restricting the angular region over which most of the radiation is directed. Thus, in order to take advantage of this increase in gain, it is necessary to point the antenna at the station which is to receive its signal. For a space vehicle system, this requires attitude control of either the vehicle itself or at least of the antenna. In practical cases, the accuracy requirements of this control vary all the way from 1 or 2 deg to 20 deg.

As was mentioned above, antennas for the free-falling body can be made extremely lightweight. However, such structures cannot withstand the forces involved in landing on the surface of the moon or re-entry into the atmosphere of a planet. Fortunately, the moon is sufficiently close that a small, fairly rigid antenna can be employed which will withstand accelerations involved in lunar soft landings. However, for planetary landings, the range is too great to permit direct transmission from the planet to earth with a small antenna capable of being brought safely through the atmosphere and capable of supporting its own weight in the planet's gravitational field after the vehicle has landed on the surface. Some other solution must be found. Two suggestions are: (1) automatically constructed rigid antennas to be deployed after the landing is made, and (2) a small rigid antenna similar to the lunar-landing antenna transmitting to a satellite relay station placed in orbit around the planet, with the relay station transmitting back to earth using a large, lightweight antenna.

These projected improvements in antenna and receiver design have been combined with the other parameters affecting communication capabilities.

The characteristics which are considered variable are listed in the left column of Table 4. Their values for the *Pioneer IV* probe are given in the next column. The third

Table 4. Communication System Capability

Characteristic	Pioneer IV	Improvement Factor, db	Extrapolated Pioneer IV	Directional-Antenna Vehicle
Transmitter power	0.16 w	+27	80 w	10 w
Vehicle-antenna gain (assumed)	1.3-2.3 db	−0.6	1.7 db	16.7 db
Ground-antenna gain	41.1 db	+4.9	46 db	46 db
Receiver sensitivity	−156.3 dbm	+7.7	−164 dbm	−164 dbm
Range	1.15 × 10^6 miles	+39	100 × 10^6 miles	100 × 10^6 miles
S/N at specified range	0 db	—	0 db	6 db

column lists a representative set of improvement factors which would make possible communication with the same bandwidths out to a distance of a hundred million miles, which is approximately the distance between the earth and Mars at the time of intersection between the probe and Mars for a shot launched in October 1960. The fourth column lists the values which would be necessary to obtain these improvement factors if the system of *Pioneer IV* were merely extrapolated to a larger size in the vehicle, and if reasonable improvements were made in the method of use for the ground antenna and in its equipment. Since the probe will be moving at very nearly sidereal rate as it approaches Mars, it will not be necessary to place the ground antenna in the automatic-track mode, but instead it may be placed on manual track, permitting an increase in antenna gain.

The last column of the Table shows what might be done if a different type of vehicle design were employed. The most important characteristic of this new design is that the vehicle is equipped with a directional antenna and has the capacity to aim this antenna toward the earth, perhaps by tracking on a radio source on earth. Comparing the Table columns for the extrapolated *Pioneer IV* and the new design, it can be seen that the existence of such an antenna on the vehicle (the antenna for this example is only 5 ft in diameter) would make possible the reduction of power from 80 watts to 10 watts. Since power is directly proportional to weight if the same lifetime is considered, a decrease by a factor of 8 in the necessary weight is implied. It should also be noted that a signal-to-noise ratio of 6 db is available for this last vehicle, whereas threshold values were considered for *Pioneer IV* and its extrapolated version. Power sources will be discussed in a separate Section.

G. Data Processing

1. Magnetic tape recording. The magnetic tape recorder has proved to be an excellent storage device for telemetered data, and it is reasonable to assume that it will continue to be the primary data-storage tool in the next five years. However, the recording of frequency-modulation telemetry with magnetic tape sometimes introduces errors into data. The principal errors are considered below in the order of importance.

The most important source of storage error is the variation of the tape speed in the magnetic recorder. Small time-scale variations of the order of 1/10 cps or more are called flutter. The lower frequency speed variations are called wow. Both variation errors are inversely proportional to the average speed at which the tape runs through the recorder. The amount of wow and flutter are also proportional to the recording frequency.

Tape-speed variations act as additional frequency modulations of the telemetry signal. When the telemetering signal is demodulated, the wow and flutter errors appear as amplitude variations of the data and usually cannot be separated out of it. Sometimes the wow and flutter reduce the threshold of the phase-lock-loop discriminators. For example, on the data record of the *Pioneer III* flight the discriminator threshold was reduced between 1 and 2 db. This effect, together with means of compensating for it, are now under investigation.

Speed variations in the tape are the result of moving parts in the recorder mechanism, including the tape itself. Capstan electricity, misaligned shafts, and bearing noise are some of the reasons for tape-speed variations. The frictional resistance of the tape moving over the guiding surfaces and the bowstring effect resulting from lengths of unsupported tape also cause speed variations. It seems unlikely that the next five years will see much in the way of reducing wow and flutter mechanically. If it is mandatory that wow and flutter be reduced, the greatest hopes lie in the direction of improving electronic compensation systems.

The second source of error is called signal dropout. This type of error is largely a function of the type of magnetic tape being used, although the tension of tape in the recorder is also a factor. Signal dropouts are likewise proportional to both the tape speed and the recording frequency.

Signal dropouts are caused by imperfections on the magnetic tape surface which lift the tape from the head and produce a momentary loss of information. Signal dropouts are also caused by the magnetic tape running askew across the reproduction head which either diminishes or completely cancels the magnetic flux in the head gap.

Errors due to signal dropouts become a problem at higher recorded frequencies (25 kc and above) and are a function of the threshold of signal-demodulation equipment.

2. Expected developments in the magnetic recording field. The maximum frequency that can be recorded with available recorders is of the order of 100 kc with a tape speed of 60 in./sec. This response is limited primarily by the head-gap width. Reducing the gap width also reduces the low-frequency response of the recorder. This is because the voltage developed by the reproducing head is proportional to the change in flux, which decreases as the gap width is reduced for low-frequency signals. One technique now being used to overcome this problem is to use a reproducing head with two gap widths, one narrow gap for the high-frequency data and a broad gap for the low frequencies. The output from each portion of the head is amplified and mixed to give an over-all frequency response from 400 cps to 1 mc at a tape speed of 120 in./sec. Utilizing this technique makes it practical to design an instrumentation recorder with an upper frequency limit at a 60-in./sec tape speed of 500 kc and, at 30 in./sec, of 250 kc. Below 30 in./sec this technique would also increase the upper frequency limit, but with present mechanical design the errors due to wow and flutter modulation would be excessive.

True video recorders of the type used to record television signals (response to 4 mc) employ a head that is mechanically rotating perpendicular to the tape motion, to increase the head-to-tape velocity. Recorders of this type could be used in the field, but would require considerable effort to maintain optimum performance. Development is now being carried on by one manufacturer to replace the moving head with a stationary head which electronically moves the point of recording across the width of the tape. This development would make the video recorder more attractive for field use.

3. Field data presentation. The amount and type of data that are presented in the individual field stations will depend greatly upon the requirements for real-time data analysis. Generally speaking, any of the types of presentations made in the final data reduction could be made in the field if there are sufficient requirements.

Each field station should contain the minimum data-display capability that would establish system performance. The resulting analog and digital records would establish system performance, and permit some field data analysis.

Analog data plotted with respect to time, on a direct-writing oscillograph, is by far the simplest form of data display. An alternate to this is a digital print-out system of the counted frequency. With good system design and proper preflight calibrations, data from these records can be read with better accuracy than the oscillograph analog method provides. The accuracy of the digital data is limited by the data frequency, counter sampling period, frequency of the subcarrier channel, and the $\pm$ 1-count accuracy limit of the counter. When the data frequency is low, such as for temperature measurements, the printing-out of subcarrier frequency can be read to an accuracy of better than 0.1%, with a 10-sec summing period. As the data frequency increases, the sampling period must be decreased to maintain this accuracy, but as the sampling period decreases, the number of significant figures read is reduced and the $\pm$ 1-count accuracy of the counter increases the error.

There are two techniques which could be used to maintain high accuracy as the data frequency increases.

The first method requires that field-station counters measure the period of frequency, and also a 10-period average. Using the low telemetering channels, below 1 kc, the 10-period average measurement will give four significant figures with a sampling time of 0.025 sec or less. The data printed out would be the inverse frequency, rather than the frequency. This method will require additional synchronization circuits when channels are measured in sequence.

The second method that could be used (Fig. 42), makes use of a phased-locked frequency multiplier. Multiplication of the discriminator VCO frequency by 10 will reduce the sampling period by one tenth and maintain the same accuracy. This method is preferable to the first one, because it can be added without changing the type of readout, and will not require added synchronization circuits.

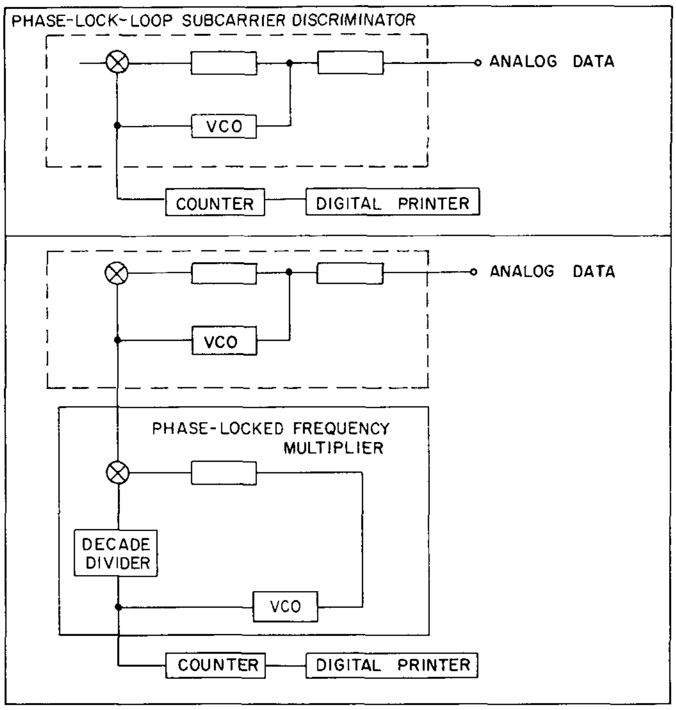


Fig. 42. Data-Handling System with Frequency Multiplier

Future development in the area of field data presentation should be directed toward the following goals: (1) conversion of digital printout of frequency to linearized engineering units with sign and decimal points, and (2) addition of the digital data to the analog record, to give one record containing good-accuracy analog and high-accuracy digital data.

4. Telemetering data reduction. It is proposed that the data-reduction philosophy developed with the *Corporal* and *Sergeant* missile systems be continued in the space program. For example, the telemetering data produced from a typical *Sergeant* missile flight would consist of:

1. Field records with about 5% accuracy and sufficient chart speed to resolve a majority of the data.

2. Quick-look data books, produced in the telemetering data laboratory, containing time-compressed composite records and individual records of each measurement with a calibration grid. These books are reproduced and distributed to all interested groups.

3. After analysis of these data is complete, requests are made to the telemetering data laboratory for high-speed records to obtain better time or frequency resolution, or to be digitized for further computer reduction.

At first thought, producing the data as described above for a space flight that might last for weeks or months seems like an impossible task, although each flight will consist of a series of phases that are easily divided, such as the launch phase, the flight through space, the entry or terminal-guidance phase, and the planet-data phase.

A space flight consisting of these phases would require a data book for each of the four phases. By far the longest period of time would be covered by the second phase, the flight through space, but the amount of data during this period would be low. The flight system should be designed to transmit only very low-frequency analog data during this phase to permit a large amount of time compression (i.e., hours of flight equal to inches of record) of the final data.

To establish a telemetering-data-handling facility capable of producing such records the following developments are needed:

1. Magnetic tape recorder with tape speeds in the order of 0.25 in./sec to record data in the phase of flight through space.

2. A fixed timing system that will generate a coded signal to be recorded on magnetic tape in the field, and upon playback will produce a wide variety of coded timing signals.

5. Final data reduction. The portion of the data-reduction problem being considered centers on the data transmitted from the scientific package itself and excludes the larger data-processing problem involving trajectory computations, rocket performance, and real-time control of the probe from the earth. It has been pointed out that certain minimum data-display capabilities at the acquisition sites are necessary to establish the system performance of the telemetering. The quick-look records assist in the acquisition of the data by answering such general questions as: Are objectives being accomplished? Is the telemetering functioning properly? Can preliminary results be stated?

The quick-look records are either analog data plotted on direct-writing oscillograph recorders or digital printouts which tabulate the telemetering subcarrier frequency vs time. In general, these forms of data recording are not suitable as final data records. It is necessary to correct the raw data; the record should be linearized, taking into account the characteristics of the measuring transducer, the telemetry link, ambient conditions in the vehicle, etc. The data should then be converted to engineering units; that is, with zero offset and scale factors applied. In more sophisticated data reduction, dynamic corrections can be applied if a knowledge of the transfer function of the transducer–telemetry system is available. After the raw data have been converted to engineering units as outlined, it is usually desirable to reduce the data even further by correlating the measurements with other pieces of information. For example, in the *Explorer* and *Pioneer* measurements of cosmic-ray intensity, a plot and tabulation of roentgens-per-hour vs position in space would be considered a suitable final form for the reduced data.

By employing modern automatic data-reduction methods the telemetered measurements from the space probes can be reduced to a presentable form rapidly and efficiently. To accomplish this goal a world-wide data-handling net is proposed as an adjunct to the world-wide tracking net.

It will be necessary to relay the data from the acquisition sites to a central data-processing center for the following reasons:

1. With many acquisition sites located around the earth, each receiving signals only part of the time, it is evident that each site is in possession of only some of the data.

2. To correlate time of reception with position in space it is necessary to have an ephemeris of the probe. The ephemeris is available only at the central tracking computer.

3. Computing facilities contemplated for the acquisition sites will not be capable of reducing the data to the most desirable form noted above.

4. If the data were fully reduced at the acquisition site, it would still be necessary to transmit the results to a central headquarters for release to the public. In many cases the results might be sufficiently momentous to warrant immediate disclosure. A radio relay is therefore called for.

6. On-site equipment and computer. The following discussion outlines a data-reduction system consisting of limited on-site computing, that is, a partial data reduction to reduce the traffic-handling requirements of the relay; a radio relay, using digital coded techniques, connecting the acquisition sites to the central computing site; a central data-reduction computer having flexible input and output equipment and of sufficient capacity and versatility to handle the vast amount of data projected in this 5-year study.

In addition to the equipment required to assure the proper functioning of the telemetering, equipment is necessary to prepare the data in a form suitable for the radio relay to the central data-reduction site. This equipment consists of electronic analog-to-digital converters, gated counters sampling the subcarrier frequency, digital time accumulators, computer-type digital tape transports, and a small general-purpose computer used for partial data reduction (see Fig. 43).

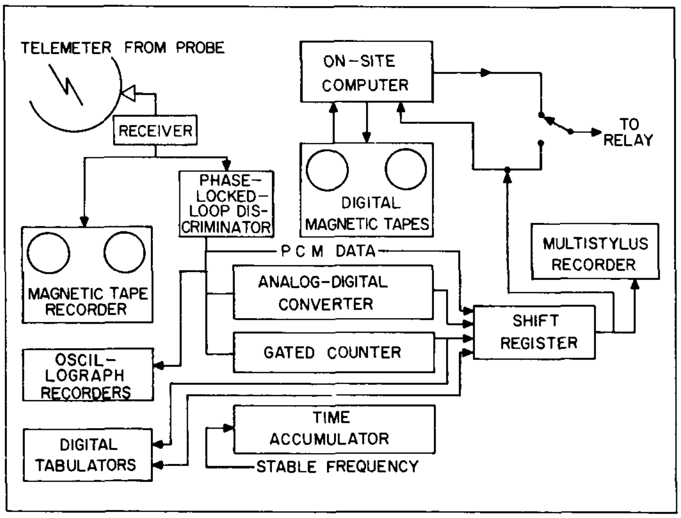
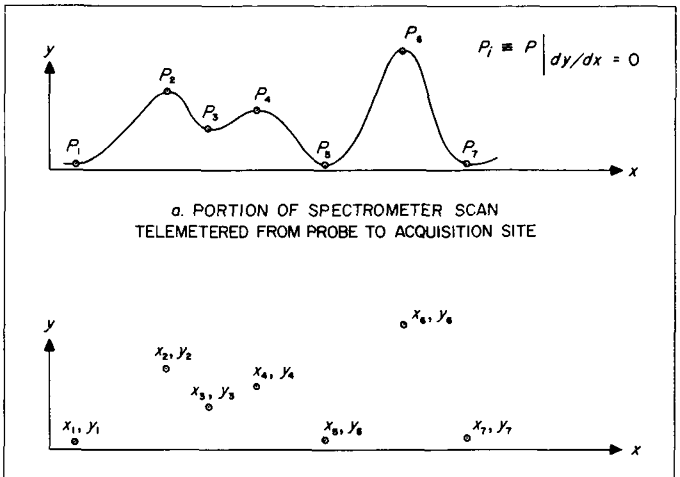


Fig. 43. On-Site Data Preparation Equipment

Commercial analog-to-digital converters are available possessing errors of less than 0.05% of full scale at sampling rates in excess of 25,000 samples per second. If required, even better equipment could be obtained by a slight advance in the state of the art. Gated counters and time accumulators are readily available and there are many digital tape transports suitable for the application contemplated. There are many computers that could serve as the on-site computer, but it would be premature to specify a machine at this time. The computer requirements should be correlated with the tracking problem; the final computer agreed upon for both tracking and data-handling requirements might be a hybrid machine. For example, some aspects of the tracking problem call for an incremental DDA computer, whereas the data-handling problem clearly calls for a general purpose arithmetic type computer. The on-site computer chosen will be a solid-state machine, allowing for transportability, minimum maintenance, and the ability to withstand extended environments. As far as data handling by the computer is concerned, it must be capable of accepting inputs from many different sources, and in different forms, and must be capable of recording its output on both magnetic tape and punched paper tape for relaying over the teletype links.

A major function of the on-site computer will be data compression; that is, the bandwidth of the data will be reduced to minimize unnecessary traffic over the relay. To state this another way, the cost and complexity of the radio relay are reduced by an increase in the cost and complexity of the data-handling equipment at the acquisition sites. An example of information of bandwidth compression is shown in Fig. 44, which represents a possible spectrometer scan telemetered from the probe. The full scan is digitized and fed to the on-site computer. The computer, in turn, picks those points on the scan where the first derivative changes sign and transmits to the central computer ordinate and abscissa values as indicated in Fig. 44.

It is obvious that such bandwidth compression is also possible from the probe itself to the earth, but whether this partial data reduction is accomplished in the probe or at the acquisition site is a simple matter of economics. It might be that the weight and complexity of the added data-reduction equipment in the probe are greater than the weight and complexity saved by using partial data reduction. By the same token, it might be argued that the use of an on-site computer at the data-acquisition point is a frivolous expense when all that is required is a simple increase in relay capabilities. Surely the problem of relaying information 100 million miles through space is more difficult than the



a. PORTION OF SPECTROMETER SCAN TELEMETERED FROM PROBE TO ACQUISITION SITE

b. INFORMATION RELAYED FROM ACQUISITION SITE TO CENTRAL COMPUTER

Fig. 44. Spectrometer Scan Showing Possible Bandwidth Compression

problem of relaying this information 5000 miles around the earth. An engineering study of the problem should be begun immediately. In any event, the use of an on-site computer will only assure the speeding up of the data flow; it will not be a necessary link in the system. If the on-site computer were to fail, the standard methods of telemetering analog recording as well as digital tape recording will be available as a back-up.

Other examples of data compression are even more obvious than the one noted above. For example, a probe on its way to Mars might be making a survey of the magnetic field between the earth and Mars. Since very little data of a startling nature could be expected over a period of many days, it is not necessary to clutter up the relay with continuous results of the survey. Rather, the on-site computer, with its digital magnetic-tape recorders as memory, can accumulate the results of a week's survey and transmit the data during a lull in traffic. As another example, when a meteorite impact is recorded, a code signifying this fact with time of reception could be relayed. This particular telemetering channel is then simply monitored at the acquisition site until another event occurs.

Relaying of the partially processed telemetered data from acquisition site to the data-processing center should, in some cases, be done as rapidly as practicable. The luxury of first inspecting the quick-look record at the acquisition site and then deciding how best to separate the data from the noise would no longer be permissible.

It might be possible to record the telemeter on a magnetic drum or disc (to avoid the problems of flutter and wow that magnetic tape possesses) and, some milliseconds after recording, to pick up the information to be fed to the discriminators and digitizers. The delay is utilized for automatically analyzing the data, determining optimum loop-filter settings, suitable sampling rates for the digitizer, etc.

The fact that the data will be relayed to the central computing site over a digital radio relay, as well as being digitized before it can be entered into the on-site computer, means that most of the data will be handled in a digital form. This method of data handling has many advantages not yet touched on. To quote Nichols and Rauch,[2] "PCM methods are ideal for relaying purposes because as long as the carrier signals are well above threshold, no noise is added by the individual links in the relay chain; the noise arises only from the quantization before the initial transmission. For the same reason PCM is ideal if the code groups are stored as such or fed into digital computing equipment, because then no additional quantization noise is introduced." Digital storage is more immune to errors of recording, storage and playback. Problems of wow and flutter in magnetic-tape storage do not exist. It is required only that the digitizing rate and quantization levels be such that the sampling and quantization errors be negligible.

With the data in digital form it will be possible to use multistylus digital recorders for quick-look viewing of the record. A multistylus digital recorder developed by Radiation, Inc. employs approximately 640 styli to record electrically on a 12-in. strip of Teledeltos paper; many such recorders were supplied for use in the *Vanguard* system. The sampled digital values are fed into the recorder and there, through a logic circuit, a single stylus is excited and a mark is recorded on the paper. In addition, at one edge of the paper, a group of styli can be reserved for recording the digital value of the record or the time in numeric form. An example of such a record is shown in Fig. 45. These recorders have an advantage in that the recording method, being purely electronic, is not troubled by limitations in recording pen, dynamics, resonant frequency, overshoot, etc.

*7. **Relay.*** It is assumed here that the relay will be an extension of the relaying equipment already being installed in the tracking network, namely, the Collins Kineplex system. In the early flights scheduled, the Kineplex system being installed will have sufficient capacity to handle most of the data; that is, a few teletype channels reserved for data relaying will be adequate. A single teletype channel transmits data at the rate of 60 words per minute (6 digits per second or 30 bits per second). It is evident that normal teletype channels would soon be overloaded by the data expected from the later experiments; therefore, it is reasonable to expect that at least a full Kineplex channel will be reserved for data relaying (see Fig. 46). A Kineplex channel has the capability of transmitting 2400 bits per second. The Collins TE-206 Kineplex system has available equipment to record and relay directly from IBM magnetic tapes, IBM punched cards, punched tapes, etc., as well as providing redundancy bits and error-correcting codes. During periods of poor transmission it may be necessary to reduce the data rate capability by one-fourth to obtain an improvement in signal-to-noise ratio. Collins accomplishes this by changing the modulation method and halving channels in a frequency-diversity system. Whether an information rate of 2400 bits per second is adequate is dependent to a large degree on the capabilities of the on-site computer. The on-site computer can also assist in the problem of data checking, helping to assure error-free relaying.

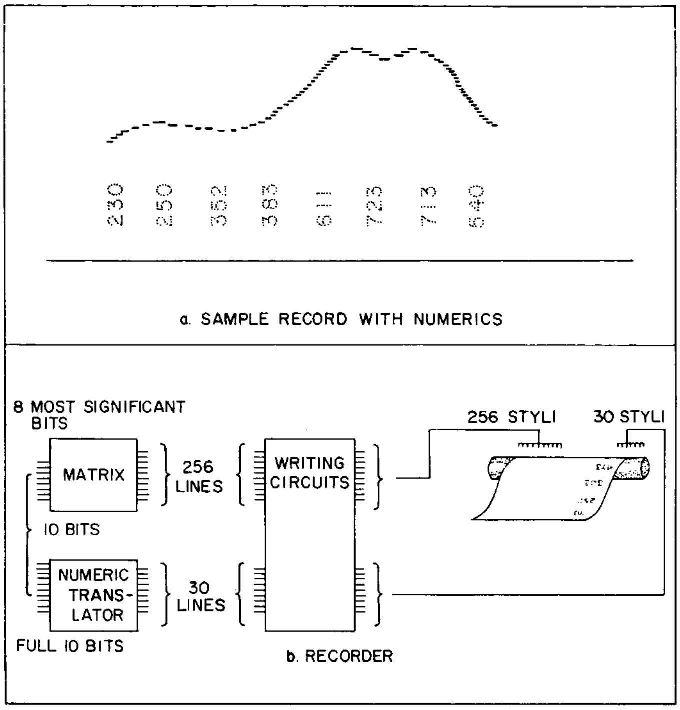


Fig. 45. Multistylus Recorder

[2]Nichols and Rauch, *Radio Telemetry*, 2nd Ed., John Wiley and Sons, Inc., New York, 1956.

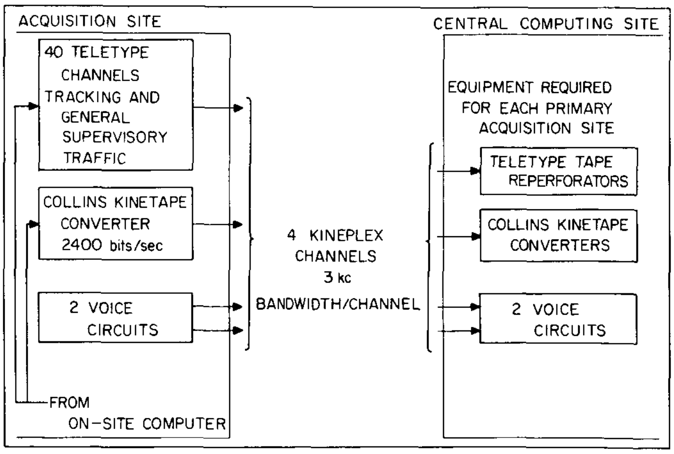


Fig. 46. Kineplex Relay System

8. Central computer. The central computer should be located near the tracking computer so that they may share equipment and personnel (see Fig. 47). The tracking computer can supply ephemerides to the data-reduction computations where correlations between measurements and position in space are required. If a tracking computer more versatile than the IBM 704 presently at JPL is obtained, such as the IBM 7090, the tracking and data-reduction computers could be the same machine, although the programming would be handled as if separate machines were being considered. New computers such as the IBM 7090 are particularly adapted to handling data-reduction problems. These computers have powerful interrupt features as well as extensive input and output capability. When used in data reduction, the computer will not be required to do very much arithmetic computing, but instead will function as a sophisticated bookkeeping machine. All of the data relayed to the computer from the many probes in flight will be assembled and stored in the computer's magnetic-tape storage. The computer's output devices feed *x-y* plotters and fast tabulators, and the significant data received are immediately presented for study. Much of the data is held in the computer's memory to be presented when more data are available so that correlations and cross-plots can be generated. Many of the output plotters of the computers will be plotting quantities such as strength of magnetic field or radiation encountered vs distance from the earth or from the planet being surveyed. In some cases it may not be possible to present an accurate picture of these quantities as plotted against the position of the space probe at the time the data are received. An accurate final trajectory might not be known until some later tracking data have been obtained. It is then possible to have the computer re-present the data in the light of more accurate knowledge.

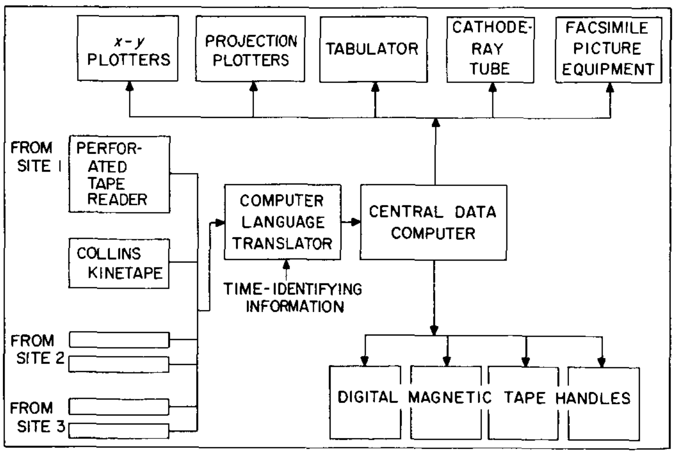


Fig. 47. Central Data-Reduction Facility

It should be pointed out that although the data are being partially reduced by the on-site computer before relaying, the small on-site computers should not be considered as arms or branches of the central computer. Their function is simply to speed the flow of data by compressing the bandwidth required for the relay, checking the data, and assisting in the control of traffic over the relay. The final data reduction and, in fact, all of the data reduction could be carried out at the central computer with little, if any, assistance from the on-site computer. As with all the ground equipment used in the space-probe program, should any portion of the system fail, the failure cannot be catastrophic and cause a shut-down of the whole experiment.

An advantage of a general-purpose data-processing computer is that the data reduction required need not be completely specified at the time the experiment is being designed. Each experiment, instead of requiring new hardware and one-of-a-kind data-reduction equipment, will be handled by modifying the computer's program. It should be pointed out that the field of data-processing digital computers is under intensive development and many new components are being made available. It is visualized by the computer manufacturers that these data-handling machines will be useful for automatic-production plant control, process control, data logging, test-facility operation, etc. It is therefore not necessary to undertake a general development program to implement the data-presentation requirements. It will be necessary only to develop special types of data-presentation equipment, such as the conversion of picture scans on facsimile machines.

In addition to the more versatile computer contemplated for JPL, there will be a need for more versatile input and output devices alluded to previously. Such an equipment is the Computer Language Translator.[3] This device takes input digital data in almost any form and at almost any rate and prepares a standard computer tape which can be handled in the computer's tape handler without tying up the main computer. The tape could also be used in an off-line printer or plotter, again without tying up the computer. Special output devices would include more rapid tabulators and automatic plotter, such as the Magnetic Tape Dataplotter.[4] This machine reads computer magnetic tape and has the ability to plot either symboled points or continuous curves. The digital computer records x- and y-coordinate numbers on one of its magnetic-tape units. The plotter then reads the x-y coordinates and shifts them into a digital-to-analog converter, which in turn drives the plotter servos. If the plotter is being utilized as a point plotter, after the servos reach their coordinate positions, a point with an appropriate symbol is printed.

For continuous plotting an ink pen is used and the digital-to-analog converter is fed, at a fast rate, with x-y coordinate numbers, e.g. 75 points per sec, the coordinate points lying quite close to each other. The outputs of the converter are then a series of small steps which to the x-y plotter appear to be continuous, slowly changing analog voltages. In addition to x-y plotters drawing curves on paper, projection plotters[5] are available in which large multicolored plots can be projected on screens; using polarized-light projectors, three-dimensional plots can be made. In addition, IBM can supply a cathode-ray output which can present plots or numerals for view or to be photographed and later printed.

9. The video problem. The problem of handling the picture data to be obtained is treated separately from the general data-reduction problem since both the picture-recording equipment and picture-transmission equipment are in a state of early development at this time, and it is quite difficult to predict their final form. Picture data differ from the telemetering already considered primarily in the fact that very much more data, many thousands more bits, are required to transmit a picture from a probe. Bandwidth-compression methods, although possible, are considered too complex for use at this time.

The following points should be emphasized: First, if the picture data are to be telemetered to the earth over a narrow-band channel, such as 8 cps, it is entirely possible to relay the telemetering information, much as the other channels will be relayed, over the digital radio relay described. That is, the picture data are first digitized and then sent to a central facility. At the central facility the information is converted back to analog form and wirephoto or facsimile equipment produces the picture. The reason for transmitting to a central site in this case is simply that the bandwidth is so narrow that the transmission can be handled. Also, if the picture is relayed over a narrow-band channel, it is entirely possible that part of the picture will be acquired at one site and the remainder of the picture recorded at another site. Second, if probe transmitter power is available, the picture should be transmitted over a wide-band channel so that a full picture could be recovered in a short time (about half an hour). This will be particularly necessary for some of the payloads where many hundreds of pictures will be required to survey the planet completely.

In this case, it is obviously desirable to place the data-reduction equipment at the acquisition site. The teletype relay could never hope to accommodate the amount of traffic required. A third possibility in picture transmission might be handled by transmitting the whole picture in one burst of information (i.e., in a few seconds). This would be the case in which, in the probe, a battery or capacitors are charged by means of solar cells, and after some period of time a scan is taken and telemetered in real time. This method avoids the necessity of including tape recorders and playback equipment in the probe. Such a possibility could be advantageous from a data-reduction standpoint if the taking and transmitting of the picture could be controlled from the earth. Then all picture data could be handled at Goldstone. If the transmission could not be triggered from the earth, all the acquisition stations would have to be ready at all times to begin receiving picture information when the probe was ready to transmit.

In any case, the handling of picture information must be implemented by the organization designing the

[3]Electronic Engineering Co., Akron, Ohio.
[4]Electronics Associates, Inc., Long Branch, N. J.
[5]Fenske, Fedrick and Miller, Inc., Los Angeles, Calif.

picture-taking apparatus. The picture-taking equipment will be tested by using the picture-reproducing equipment developed in conjunction with it.

H. The Tracking of Space Probes

The determination of the flight path of space probes has at least four purposes which, for convenience, are placed in the following categories: (1) prediction of angular positions, angular rates, range, and range rates with respect to observation stations, (2) measurement of deviation from the preflight nominal trajectory for use in computations related to corrective maneuvering of the probe, (3) correlation of probe position and attitude with scientific measurements, and (4) evaluation of missile, guidance, and communication systems.

It should be noted that only the first two categories require real-time transmission and reduction of tracking data.

The motion of the center of gravity of a body within the solar system, in the presence of gravitational forces only, can be described completely by specifying at one time the exact position and velocity vectors with respect to the sun and the vector sum of the gravitational forces. If it is assumed that the motion of the earth with respect to the sun is known, specifying the position and velocity vectors with respect to the earth can be made to be equivalent.

The essential element of tracking is the obtaining of measurements of sufficient number and quality for determining the required six parameters. If radio-tracking devices are used, range, rate of change of the range, and two angles may be measured. If optical telescopes are used, two angles can be measured. With neither type of instrument is it presently feasible to measure angular rates. Except in certain geometrically degenerate cases, it is possible to determine the six desired parameters from any six measurements if they are made with infinite precision. In any realizable situation a redundancy of measurements is required to obtain the desired accuracy of flight-path representation.

The difficulties encountered in determining the position of bodies moving in almost uniform gravitational fields using only angular measurements are well documented. For example, the absence of a reference dimension of adequate accuracy causes significant uncertainty in the determination by astronomical means alone of the fundamental astronomical unit, the mean earth–sun distance. It cannot be hoped that angular measurements of space probes far from the earth can be made more accurately than measurements of planetary motions made by the astronomical community in the past centuries. If the angular motion of a probe is measured only far from the earth in an almost uniform gravity field, the determination of range will be nearly impossible using angular data alone. However, near the earth, or whenever astronomical bodies are approached so closely that the effects of the variations in the gravitational acceleration with range from the encountered body can be measured, the ability to determine range from angular measurements alone is significantly improved.

The reduction of tracking data is essentially the problem of filtering, by statistical analysis, the random observational errors and the systematic bias errors. At JPL, a computational program was constructed for the tracking of space probes which utilized an IBM 704 electronic computer. The program was most recently used during the tracking of *Pioneer IV* and can be considered as a prototype of more advanced computational schemes.

The basic procedure is as follows. A set of initial conditions is assumed or obtained from iterating within the program and used to start the integration of the exact drag-free equations of motion. The computed trajectory variables are transformed into station-referenced coordinates and corrected for refraction and station anomalies. The differences between computed and observed values are used to determine those corrections in initial conditions which result in the minimum sum of squares of the differences between calculation and observations. The corrections in initial conditions are added to the previously employed initial conditions and this completes one iteration.

Figure 48 is a block diagram of the trajectory-computation program. The initial conditions at time of injection are assumed or specified from the tracking program. The input coordinates are distance from the earth's center, geocentric latitude, longitude, and the magnitude, elevation, and azimuth of the velocity relative to the earth. The integration of the trajectory is carried out in a right-handed, earth-fixed, space-oriented Cartesian coordinate system where X is the direction of the vernal equinox and Z is the direction of the north polar axis.

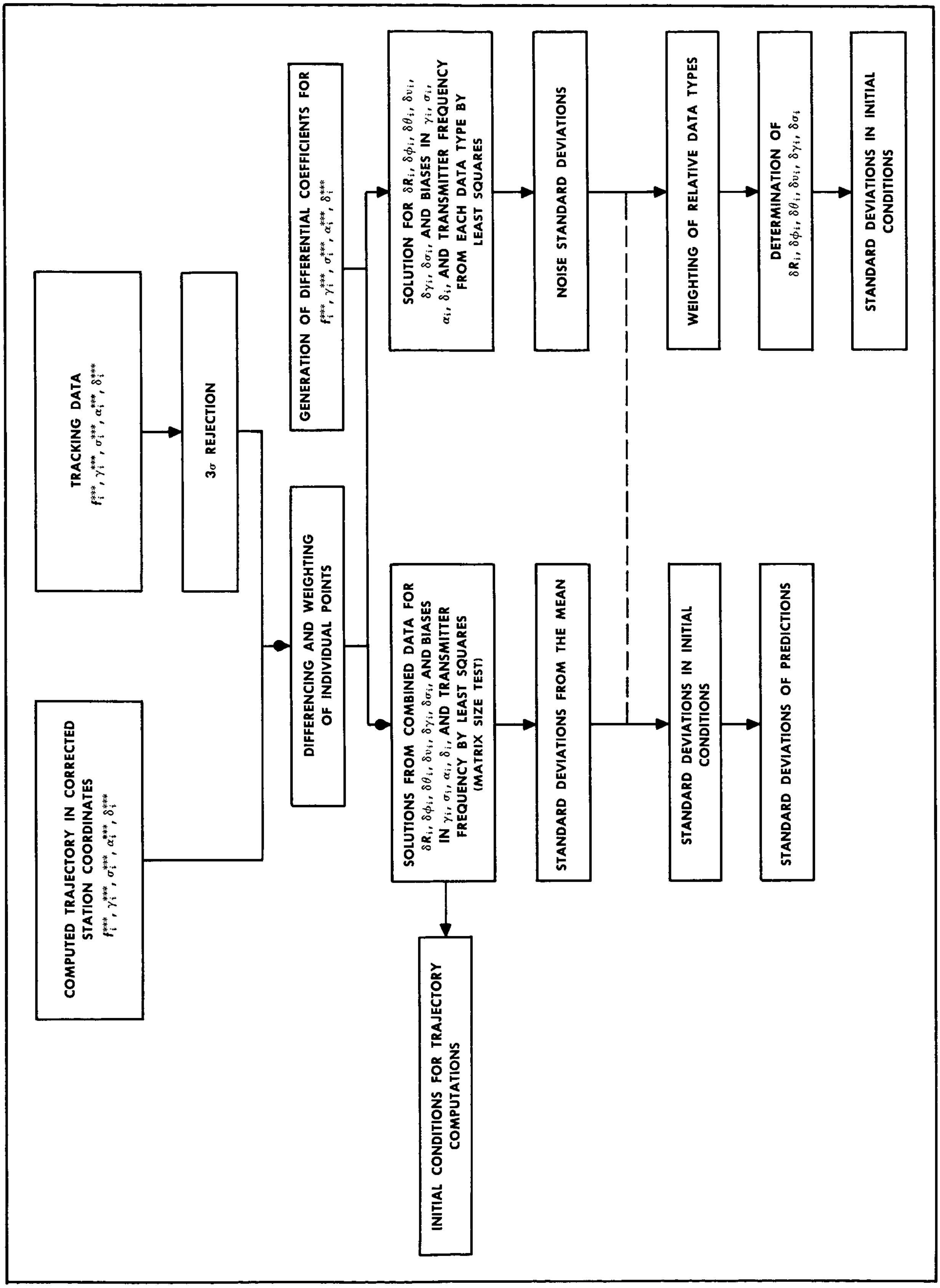


Fig. 48. Trajectory Computation Program

For purposes of the tracking program, the trajectory computations are provided in terms of coordinates as similar as possible to those being observed at the tracking stations. The radial rate is converted to doppler frequency and then scaled and biased corresponding to the way in which the individual stations are mechanized. Angle data, corrected for refraction, are provided in elevation–azimuth and local-hour-angle local-declination coordinate systems. In addition, probe position can be displayed in geocentric, geomagnetic and target-oriented coordinate systems. A variety of angles involving the direction of the angular-momentum vector of a spinning payload and distances, velocities, and directions between various bodies and observational stations is also printed out.

The tracking program shown in Fig. 49 accepts as inputs the data obtained from the tracking station and the computed values of the estimated trajectory in terms of the coordinates measured at the stations. Prior to full acceptance of a data point into the tracking program, the difference between the computed and observed values is compared with a standard deviation, which is either an externally specified number or one computed within the tracking program from earlier observation points. Measured values which differ from the computed values by more than three times the standard deviation are rejected.

Individual data points are weighted inversely as the variance of the deterioration in quality of the tracking data. The weighting used may depend on whether an automatic tracking mode is used, on the signal strength, and on the elevation angle.

Differential coefficients are at present computed by differencing six trajectories with perturbed initial conditions from a reference trajectory. Since the reference trajectory does not need to be the same as that used for predictions, differential coefficients need not be continuously recomputed.

The differences between computations and observations, properly weighted, and the differential coefficients of the observations with respect to the initial conditions are fed into a number of least-squares-fitting routines. In the primary method, each data type is weighted inversely as the previously computed standard deviations from the mean for that type and then is combined. Changes in the six initial conditions and in constant biases in the five possible observation types can be solved for. Thus, the maximum matrix size provided for is 11×11. The matrix size test which can, on option, presently be used is based on the ratio of the changes in initial conditions and biases called for and the computed standard deviation in initial conditions. The new initial conditions and biases obtained by adding the changes solved for are used as input for subsequent trajectory computations. Standard deviations from the mean and standard deviations in initial conditions are always displayed. Standard deviations of predictions are computed on option.

Least-squares routines are applied to each separate data type in order to obtain the changes in initial conditions called for by the various types. The initial conditions so solved for are used to obtain standard deviation from the mean for the optimum fit to each data type separately, and are called "noise" standard deviations. The separate changes in initial conditions obtained above are combined by weighting the results from each type inversely as the variance of that data type from the pointing trajectory. Standard deviations of the initial conditions obtained in this manner are also computed.

If radio tracking devices are used, the accuracy of range and range-rate determination is determined principally by the stability of the oscillator involved. It can be stated that no presently available oscillator suitable for payload use has sufficient stability for use as a primary source of tracking data. The oscillators used in interrogating units for transponded systems should prove of adequate stability.

With respect to the angular accuracy attainable with radio telescopes, two factors appear to be important: (1) the magnitude of random errors (and to some extent the validity of assuming them to be gaussian) and the degree to which they can be represented by a homoscedastic set in the presence of variations in signal strength and geometry with respect to the horizon, assuming that the effects of winds are determinable and all other effects are repeatable, and (2) the stability with time of bias errors and the extent to which the variation of these biases with signal strength and geometry is like that measured during preflight calibration.

The antenna at Goldstone Lake used by JPL during the tracking of *Pioneer IV* was evaluated using in part the tracking-computing program described above. The data were found to have a standard deviation of about 1

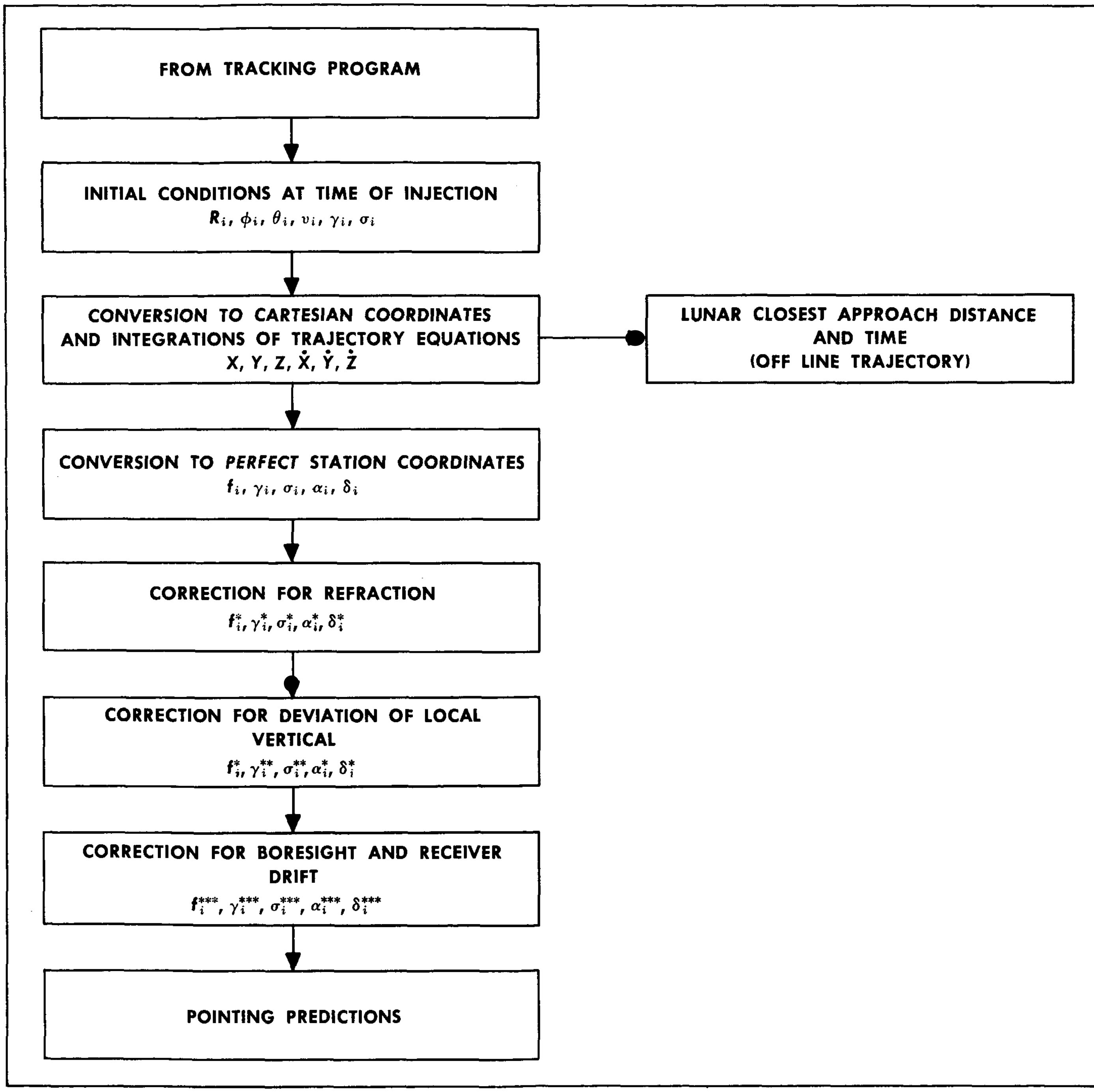


Fig. 49. Space Probe Tracking Program

minute of arc which, because of the large quantities of data obtained, resulted in errors of the mean smaller by 1 or 2 orders of magnitude from this source. The uncertainty in the determination of biases appears to be about 1 minute of arc with present data-reduction procedures. Computational methods for further improving the estimation of station biases are under construction and should soon be available.

I. Space Power

The auxiliary power system is required to provide all the probe power at the desired voltages and frequencies. Analysis of the space power problem reveals four major technical subdivisions: energy sources, energy conversion, control systems for energy conversion, and solid-state regulated ac and dc power.

The key fact regarding space power is that the total energy requirements become so large over the long operating times that radiant-energy converters and nuclear-energy sources are much lighter in weight than conventional chemical and electrochemical energy sources. Any conventional energy storage system (electrochemical batteries, fuel cells, chemical propellants, etc.) becomes excessively heavy as the operating times exceed 50 hours, because the weight of the system increases directly with the product of power level and operating time. For radiation-energy converters (solar cells) and nuclear-energy sources (radioisotope thermionic diodes and reactor-turbo-alternators), weight increases as a function of power level only. This "power level only" factor gives solar and nuclear power a tremendous weight advantage for both instrumented and manned space probes. For manned space flight the required power levels will probably rise above 5 kw.

The following tabulation presents the watt/lb performance of various systems based on a 50-hour operating time:

Electrochemical batteries and fuel cells	1.0 to 3.5
Solar cells at earth	5.0 to 12.0 (no protective cover)
Radioisotope thermionic diodes	8.0 to 15.0 (1 watt to 100 watts)
Radioisotope thermoelectric semiconductors	5.0 to 15.0 (1 watt to 1 kw)
Nuclear reactor-turbo-alternator	6.7 at 3 kw 11.3 at 10 kw 20.0 at 30 kw

The weight of the auxiliary control and the electrical systems associated with the various energy sources is completely overshadowed by the dominating watt-hour power requirement. Energy conversion control involves the development and design of voltage and frequency reference elements, transistor-magnetic servoamplifiers, and actuators for positioning such equipment as solar-cell panels by momentum interchange and radiation pressure. Great emphasis must be placed on reliability and simplicity.

As a result of the continuing analytical and experimental work, definitive knowledge has been obtained regarding the estimated future power requirements for space probes. Experience in the fabrication of early satellites has clearly demonstrated the necessity for close integration of the space power system with the payload. The interrelated positioning problems of the solar cells, the antenna, and the optical head are typical.

As indicated in the previous Section, strong efforts have been made to minimize the average power demand of the space probe by using advanced radio communication techniques and by transistorizing all the scientific experiment and guidance equipment. For both lunar and interplanetary (e.g., Mars) flights, the long operating times of 75 hours and 150 days, respectively, place energy storage power sources such as electrochemical batteries or monopropellants at a great disadvantage to solar cells and nuclear energy sources.

The average power required for the lunar space probes now being planned for the next few years varies from 2.5 to 2000 watts. For typical lunar missions a power pulse of 200 watts is required for 3 hours after take-off for lunar acquisition and again at moon intercept for terminal guidance. An estimated additional 20 watts for 5 hours is required to transmit a lunar picture to the earth. Preliminary models of solar cells and radioisotope thermionic diodes are under evaluation for use as the power source. Energy stored in electrochemical batteries (nickel–cadmium or silver–zinc) will provide the two power pulses of 200 watts for 3 hours each. One transistor-magnetic static inverter will provide 400-cps power, both square wave and sine wave, at 26 volts rms, as well as 6.3 volts dc, to power the scientific experiment and guidance equipment.

For the Mars and other planetary probes, an average power of 100 to 2000 watts is required. As in the case of lunar space probes, power pulses of a few hours' duration will be required during both the initial acquisition and terminal guidance phases. Again, as in the lunar probe, these multiple-hour power pulses will be provided by electrochemical batteries recharged during flight. One static inverter will provide all the regulated ac and dc power in order to achieve simplicity, reliability, and minimum weight. Since Mars is roughly 52% farther from the sun than is the earth, the power received by radiation from the sun is decreased to 43% per unit area of that received at the earth; and radioisotope thermionic diodes and thermoelectric elements are correspondingly more attractive than solar cells. Only nuclear-energy sources are feasible for powering flights in the solar system beyond Mars, because the solar-energy level decreases as the square of the distance from the sun. Analysis has shown that radioisotopes will be made available for space probes and shadow shielding provided to reduce the radiation background count to 0.005 milliroentgens per hour so as not to interfere with any cosmic-ray and gamma-ray experiments. These shielded-radioisotope thermionic diodes may weigh less than solar cells, and because of their small size and rugged cylindrical tungsten construction, would not be as likely to be damaged by micrometeorite showers.

1. Solar power. The silicon photo-voltaic cell assembly consists basically of a multiple array of individual cells made from a large single crystal of silicon. Hyperpure silicon is melted in a controlled atmosphere at approximately 2000°F. A minute quantity of pure arsenic is added, thus converting the melt to *N*-type silicon. A crystal of *N*-type silicon will grow on a seed of pure silicon under carefully controlled temperature and withdrawal rate.

The crystal is then cut transversely by diamond saws into wafers approximately 0.016 in. in thickness. The wafers are heated in an electric furnace to a point below their melting temperature. At this time, boron trichloride gas is introduced into the furnace, reacting with the silicon to form a thin layer of boron, and a diffusion into the silicon of boron atoms occurs, thus converting this region to *P*-type silicon and providing the necessary discrete junction with the underlying *N*-type silicon.

When photons emitted from the sun impinge upon the surface of the silicon converter, they are absorbed, with the resultant creation of electron–hole pairs. Those that are within diffusion length of the junction drift into the junction, where they are separated by the potential difference. The holes and electrons are forced to the appropriate electrode, resulting in current flow.

It has been planned to use silicon photo-voltaic solar cells that are made by doping sliced silicon crystals by boron diffusion. These cells are 0.020 in. thick and are mounted on a 0.090-in.-thick aluminum plate. The average efficiency of these solar cell packages incorporating a diode will be 7.5%. The radiant energy received from the sun above the atmosphere in the vicinity of the earth is 130 watts/sq ft. At Mars, the solar-radiation energy level is 58 watts/sq ft; at Venus, it is 247 watts/sq ft. A surface area of 2 sq ft is required near the earth to produce 20 watts when the sun's rays are perpendicular to the solar cells. The output of the solar cells falls off as a function of the cosine of the angle of the solar radiation relative to the solar cells. Output power also decreases as the temperature of the cells rises. In addition, to obtain optimum energy conversion the electrical load impedance must be carefully matched to the impedance of the solar cells. It is estimated that, during the 3-day lunar-probe travel time, the incidence angle of the sun's radiation will not vary by more than ±20 deg, which will cause a change in the output of the solar cells of not more than 6%.

The weight of the solar cells and their supporting structure was planned to be approximately 0.013 lb/sq in. The weight of a 20-watt lunar-probe solar-cell panel would be slightly less than 4 lb.

Solar-power-panel design is determined by the power output required, the solar-radiation flux level, the angle between the center line of the payload and solar radiation during flight, and the attitude control of the payload. The solar cells would be mounted either directly on the payload structure or on panels positioned so as to achieve as nearly as possible 90-deg incidence to the sun's rays. For lunar missions it is planned that the payload attitude-control system roll-control the payload with respect to the sun's position during the 40-hour low-power flight phase so as to achieve maximum solar-radiation energy conversion. The payload would track the moon in pitch and yaw and the sun in roll (Fig. 50). The physical configuration of the payload structure would permit the solar cells to be located on any portion of the circumference or aft flat section of the payload. It would be necessary to place the solar cells in the correct position on the circumference of the payload so that they would receive maximum illumination while the moon is tracked in pitch and yaw. The position on the circumference would depend on the angle of the trajectory with respect to solar radiation.

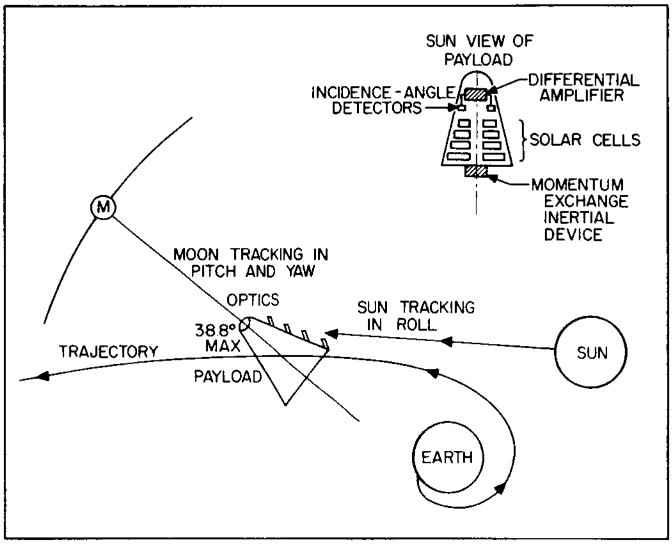


Fig. 50. Solar-Cell Roll-Control Sun-Seeker

The sun orientation of a separate solar panel would be controlled by means of momentum-exchange servo motors. A solar cell would be placed on each of the four corners of the cell package and their outputs compared. Any change in these outputs resulting from a displacement of the panel with respect to the angle made with the sun would cause an electrical signal to be delivered to the brake of the appropriate momentum-exchange motor. This braking action would result in a corrective moment of the motor housing, causing the panel to assume the original angle with the sun. An induction motor has been tested for a considerable period in vacuum (2×10^{-5} mm of Hg) to investigate the problem of bearing-grease retention and lubrication.

For operation of the payload electrical equipment during the period when the vehicle is in the shadow of the earth, a silver–zinc or nickel–cadmium battery would be utilized. The power transfer from the battery to solar cells would be accomplished by a mechanical or solid-state power-transfer switch. The output of the solar-cell module would be compared to a zener reference diode, and when the open-circuit voltage indicates full sunlight exposure, the transfer switch would be actuated.

Although the solar-cell sun-seeker roll-control system is relatively simple, it would be eliminated if at all possible in order to achieve maximum reliability. A detailed study of the trajectory, solar-radiation incidence angles, solar-panel-unfolding complexities, and solar-panel–payload-shadow and thermal-control problems of each mission is necessary to determine whether the sun seeker is required.

It is necessary to conduct thermal-radiation-balance analyses in the vicinity of each planet to determine operating temperature and efficiency of the solar-cell panels. A servo thermal-balance system may be required.

For the lunar "flyby" and landing configurations the solar cells would be designed to deliver an average of 20 watts of power, which is in excess of the average power requirements during the 60 hours of flight from the earth to the moon. During this period the solar cells may charge batteries, which would then deliver a pulse of 25 v-amp for 3 hours for terminal guidance and 10 v-amp for 10 hours for picture transmission.

On the lunar orbiter shot it would be desirable to have the solar cells provide sufficient power to operate the transmitter and the receiver as well as the scientific experiment equipment. Once the payload is placed in a lunar orbit, operation of the gyros is not required. The steering-jet solar-cell roll-control sun seeker would be inoperative because it would have run out of helium gas. For a lunar orbit the lightest system would position the solar cells at 90 deg to the sun by means of a gimbaled structure, as shown in Fig. 51. The differential output of the solar-cell quadrants would operate the gimbal-torque-drive motors in an off-on manner to hold the radiation incidence angle to 90 ± 3 deg. The gimbaled solar-cell plate would be positioned out away from the payload.

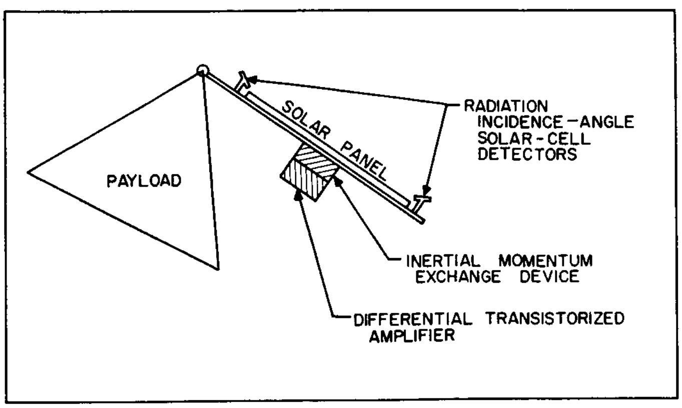


Fig. 51. Lunar-Orbit Sun-Tracking Gimbaled Solar-Cell Assembly

A problem arises in preventing the solar-cell panel from being in the shadow of the payload. Omnidirectional solar cells would be mounted on both the payload and the solar-cell-plate structure. A differential amplifier would detect the fact that one structure was in the shadow of the other structure and would slowly rotate the solar-cell panel until both it and the payload were receiving equal radiation from the sun. Of course, the solar cells would provide the energy to drive the on-off motors to track the sun as well as charge the batteries for dark-side operation and power the scientific experiment equipment and the transmitter-receiver. Signal Corps tests have demonstrated that hermetically sealed rechargeable nickel–cadmium batteries have a charge-discharge life in excess of 5,000 cycles. Lunar orbit time is estimated to be 3 hours, which would give the combination of solar cell and nickel–cadmium battery an operating time in excess of 600 days.

Solar-cell-module development work has been accomplished, including short-circuit current-matching techniques for individual 1- × 2-cm cells to achieve a maximum of 28-volt module efficiency. An artificial sun source which results in a solar-energy conversion efficiency for silicon cells which is approximately the same as the sun has been partially developed. Extensive tests have been conducted with a number of artificial sources and filters of different types in an effort to develop a combination of an artificial source and a filter that would closely match the sun's solar spectrum and total radiant energy.

Techniques are under development for mounting solar cells on minimum-weight panels that will withstand the vibration and shock environment during take-off and retro-rocket operation. Epoxy mounting of solar cells on honeycomb or waffle-tapered beam-reinforced structures appears practicable.

Preliminary design has been initiated to compare the total system weight, complexity, and reliability of various types of sun-seeker solar-panel orientation servos. The output of solar cells decreases with the cosine of the incidence angle of solar radiation. Therefore, sun-seeker servos are required, as shown in Fig. 52.

Silicon solar cells are expensive—even those whose efficiency is only 8%. Under optimum conditions at the earth, solar cells that will provide 1 watt of power cost $350.00.

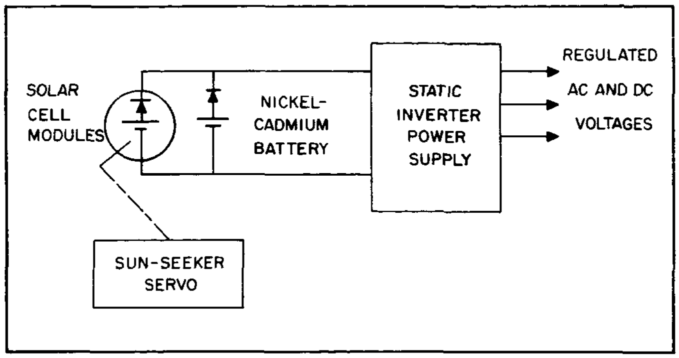


Fig. 52. Solar Power System

One of the solar power problems not yet well defined is the need for protection from micrometeorites. Space-flight tests of bare, unprotected solar cells are planned in the near future by other organizations. Informal technical liaison has been established so that the results of these unprotected-solar-cell space-flight tests will be made available for future planning. Micrometeorite impact information from the *Vanguard* and *Explorer* projects indicates that solar-cell protection is probably not required.

In addition to continuing the previously mentioned development work, the development of sandwich-type solar cells will be undertaken to increase solar-energy conversion efficiency and output per unit area. Analytical and experimental evaluation will be made of the combination of solar cells covering a thermoelectric semiconductor layer that will convert to additional electrical energy the heat generated from the 10% efficient conversion process of the solar cells.

The combination of solar mirrors and boilers with thermionic diodes, thermoelectric semiconductors, and mercury-vapor turbo-alternators has been considered briefly. If the required power can be achieved with systems that offer the same or better power per pound and are simpler (that is, do not require a liquid metal or vapor heat exchanges) these more reliable systems (solar cells) should be used.

2. *Electrochemical batteries.* Injection-guidance power —500 watts for 10 min—will be provided by manually activated silver–zinc batteries. Regulated dc and ac voltages will be supplied by a silicon semiconductor–magnetic inverter. A 500-watt battery will weigh 6 lb and have a volume of 100 cu in; a battery of this size will have a rating of approximately 19 watt-hour/lb under room ambient conditions and 17.3 watt-hour/lb after a 24-hour soak at 130°F. These data are from preliminary tests made on a prototype battery in the laboratory and could be improved upon somewhat.

Electrochemical batteries would also be flown to provide payload power for gyro operation during the lunar acquisition phase shortly after takeoff and for terminal guidance. For Venus soft-landing missions, nickel–cadmium batteries floated across solar cells could be used to deliver a power pulse to transmit information after going under the clouds of Venus where solar cells would be inoperative. The batteries would be charged prior to takeoff and would be recharged by the solar cells or the radioisotope thermionic diode during the long low-power-demand flight phase. Rechargeable mercuric oxide, nickel–cadmium, and silver–zinc batteries were considered for this application. Silver–zinc batteries that had the ability to go through one charge-and-discharge cycle reliably appeared to be quite promising because they have a substantially higher watt-hour-per-pound performance than nickel–cadmium batteries. It appeared likely that mercury batteries would be ruled out for this application on a reliability basis because of their low-drain discharge rate. It would take approximately 100 of the 40-milliwatt 1R-size mercury batteries to deliver this power. It would take roughly 30 of the largest 42R nonrechargeable mercury batteries to deliver power at the 45-watt rate required for the two power pulses described above.

A battery survey has been made of the most promising types. Environmental evaluation tests have been conducted with emphasis on the long activated stand time in a high vacuum. The U.S. Army Signal Corps has conducted a thorough environmental evaluation test on rechargeable batteries for the solar-cell–rechargeable-battery satellite application; and the results of their experience and work would be utilized in selecting the battery for the lunar-satellite flights.

3. *Nuclear power.*

Radioisotope thermionic diode. The radioisotope thermionic diode (RTD) is a heat-to-electricity energy conversion device. It will ultimately utilize the heat from a decaying fission-product radioisotope to heat the cathode, replacing the electrical heating now being used in development testing. It will then be a self-contained primary source of electrical power.

The RTD utilizes the principle of thermionic emission from a hot cathode to a colder anode, with extremely small spacing between the two. The effect of space charge is thus minimized, a voltage is established across the electrodes, and current is made to flow through an external load. High-temperature operation of a low-work-function cathode is essential.

A unit the size of a large fountain pen, exclusive of radiation shielding, is envisioned. A power range of 8 to 15 watts per unit for a 1-year period is considered feasible. The length of time of delivery of useful power is a function of the half-life and specific power of the radioisotope used. The most promising radioisotope is cerium-144, which has a 290-day half-life; i.e., at that time the power output would be one-half of its initial value. It has a specific power of 7.23 thermal watts per gram in its attainable and usable form, the compound cerium oxide (CeO_2). By contrast, the pure isotope has a specific power of 24.5 thermal watts per gram, but it is not in a usable form. The selection of this isotope is also dependent upon isotopic source strength, availability, cost, high-energy decay products, and shielding required.

The use of reactor-produced radioisotopes may also be possible, although certain limitations make their use unlikely. Alternately, solar energy or a nuclear reactor may serve as a heat source.

Electrical heating is employed instead of radioisotopes in RTD tests at present. Past efforts have established the feasibility of obtaining power from thermionic diodes, provided low-work-function materials are used, and have demonstrated the possibility of achieving sufficiently high temperatures, high vacuum, minimum plate spacing, etc.

To date, JPL efforts have been concentrated on using flat plates of large areas (0.75 sq in.) as contrasted with the use of small areas elsewhere (0.2 sq in.). The "thermo-electron engine" is shown in Fig. 53. Initial operation of the engine has been made with cathode temperatures up to 1150°C. At a cathode temperature of 1125°C and an equilibrium temperature of 900°C, 0.004 watt of power was produced at 0.2-volt potential. The maximum critical spacing was found to be 0.002 in. Power output in this fractional-wattage range is obviously of no interest; however, it was found that outputs of several integral watts at potentials in the range of 0.5 to 1.0 volts can be realized with cathode temperatures of approximately 1300°C. The

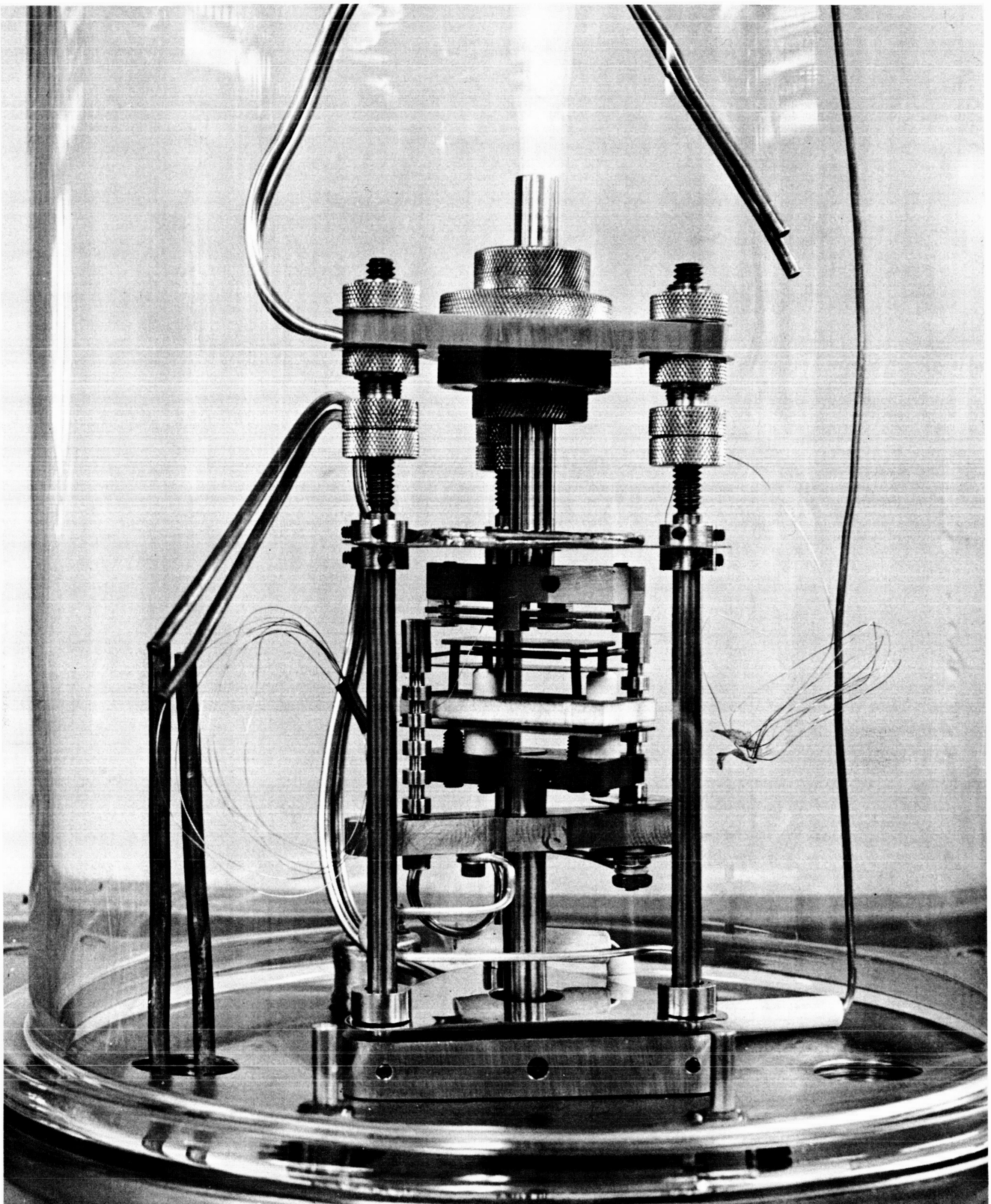


Fig. 53. Tapered Concentric Cylinder Thermionic Diode

1 COOLING WATER CIRCUIT AND COILS
2 CATHODE HEATER FILAMENT LEAD
3 ANODE SUPPORT BRACKET
4 FINNED ANODE SHEATH
5 ANODE HEATER FILAMENT
6 CATHODE HEATER FILAMENT
7 CATHODE (HOT EMITTER)
8 ANODE (RECEIVER)
9 CATHODE SUPPORT
10 CATHODE SUPPORT BRACKET
11 VERTICAL ADJUSTMENT YOKE
12 VERTICAL ADJUSTMENT GUIDE
13 TOP SUPPORT PLATE
14 POSITIONING RACEWAYS
15 MICROMETER SCREW THREAD
16 VERTICAL ADJUSTMENT SCALE

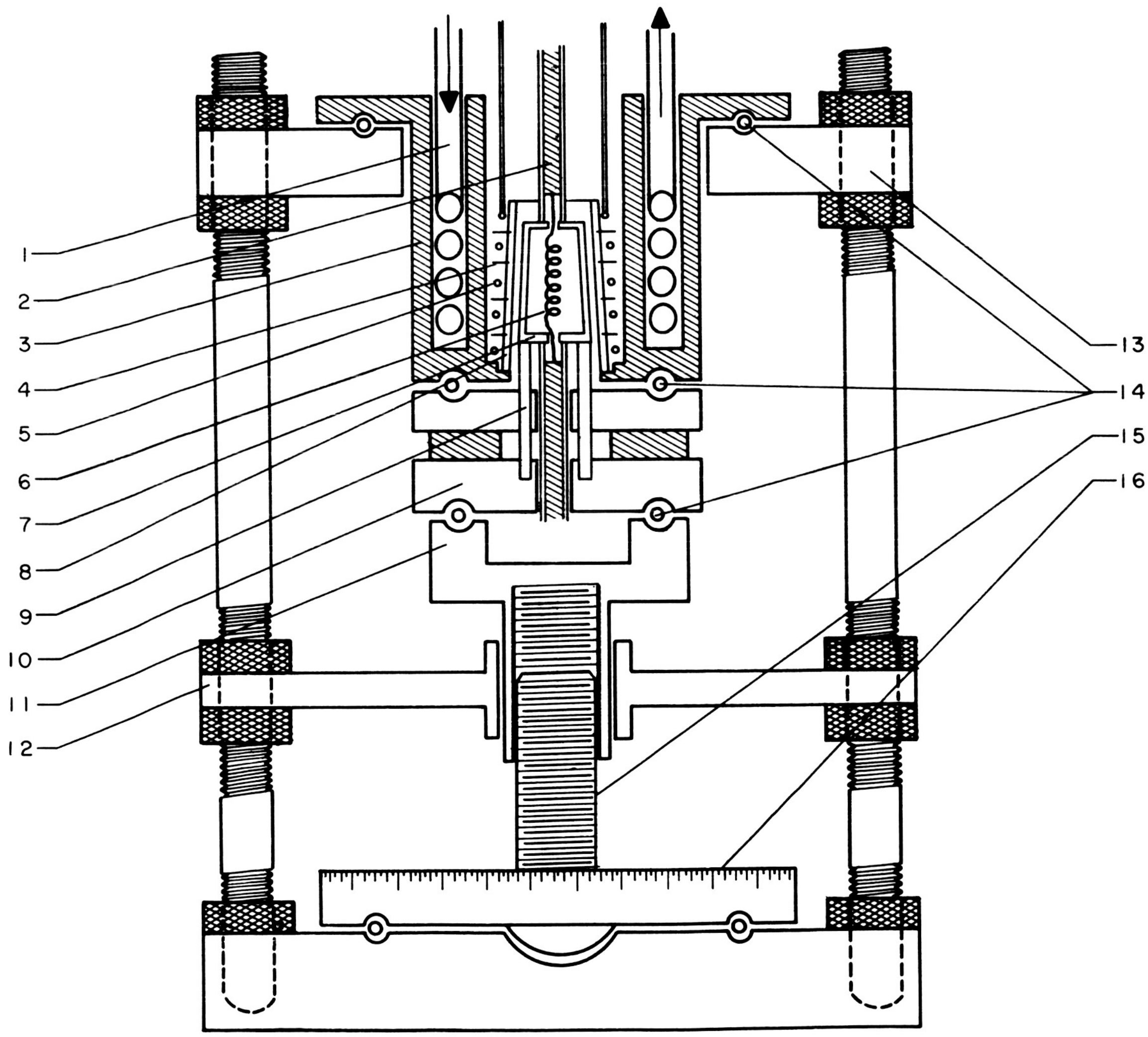


Fig. 54. Preliminary Design of Thermionic Diode

apparatus is presently being modified for operation at temperatures up to approximately 1500°C.

Concurrently, a new engine is under design using tapered concentric cylinders instead of the flat-plate geometry (Fig. 54). Completion of detailed design is anticipated by early spring 1959, followed by immediate fabrication and subsequent testing.

Thermoelectricity. The utilization of thermoelectric units as power sources is based upon the Seebeck effect, wherein a voltage is developed in a loop containing two dissimilar metals when the two junctions are maintained at different temperatures. Recent advances in solid-state physics, especially with semiconductor materials, together with the work of the Russian physicist A. F. Ioffe, have stimulated new interest in this phenomenon for power generation.

Any heat supply which will heat the hot junction to the desired temperature may be used; thus the use of radioisotopes, reactors, or solar energy may be possible.

4. Solid-state regulated ac and dc power. Nuclear and solar energy converters deliver unregulated dc electrical power. Solid-state static inverters convert unregulated dc power to regulated ac and dc power. Zener diodes and tuning forks provide the voltage and frequency reference elements. Static inverters of many different types have been developed utilizing germanium power transistors operating in a switching mode. The initial design and experimental work has begun to develop circuit techniques for utilizing silicon solid-state devices in space-power supplies. These advanced circuit-design techniques promise significant reductions in weight for both dc and ac sine-wave power supplies.

IV. PUBLIC REACTION

"Catch a falling Sputnik
Put it in a match box
And send it to the U. S. A."

. . . Sung by British school children in 1958 to a popular tune.

The entire world has come to see itself, for better or worse, at the beginning of the "Space Age." Because of international competition, a universal layman's psychology that space exploration represents "comparative prowess" in all technical areas has also come into being. The word "space" itself contains overtones of group survival and implications of cultural change.

The United States is superior to the Soviet Union in most areas of science and engineering; however, it is clear that the Soviet Union intends to influence world opinion so that American capabilities are judged by the performance of American space technicians. This places a weight of responsibility upon the space flight community which it has not solicited and which it has only begun to recognize. Corroborating this is the bald statement in a pamphlet distributed at the Soviet Union exhibit at the Brussels International Exposition: "And finally, the *Sputniks,* the fruit of the creative thought and hard work of our people, bear testimony to the technological maturity of Soviet industry. U. S. scientists are now declaring for all to hear that in the field of science the U.S.S.R. has already outstripped the United States." (See also the Public Reaction Addendum for a Soviet Union version of American space efforts.)

Abstractly, the scientific community and the lay public are in complete agreement. Both agree that nothing would be more exciting than the detection of life on one of the other planets. And they share an enthusiasm for learning the origin of the solar system. It is the manner in which they want the space technicians to implement the space exploration program that their interests do not run parallel.

One peculiar difference in the two groups is that the public has a great general interest in space flight, engendered by the comic strips and science fiction, whereas many scientists remain quite detached from it. In contrast to their keen interest in the act of space voyaging, the public is apt to ignore the more subtle problems such as planetary contamination. Their interest in these specific problems becomes aroused if some distinguished scientist makes a strong and controversial statement on the matter. But the public is likely to become enmeshed in the semantics of statement and be easily misled; whereupon the scientists become frustrated.

The layman prefers to hear his science simply stated and concretely illustrated with packaged items; however, at the same time he has a mystical faith in the abstract. The one allows him to participate and pass judgment; the other gives him an opportunity to be in awe. The cautious, carefully delineated statements that scientists prefer to make are no more satisfying to the public than general vagaries (typical of most political pronouncements) are to the scientists.

Sputnik I was a humiliating experience for many Americans; therefore, the competitive aspects of space voyages cannot be avoided entirely. Even the space technicians and foreign affairs experts who long to be free of the competition will not feel at ease until the United States has recorded a "first." Space spectaculars, failures and successes, will become another anxiety of the American scene.

While the ultimate value of space exploration depends upon the intrinsic merit of the scientific experiments it performs, the success of the experiments in turn depends upon rocketry know-how and the public's willingness and ability to finance its implementation. The willingness of a free and honestly informed community to finance space science depends upon its collective curiosity and its pride, accompanied by a sense of urgency to satisfy them both.

Most scientific experiments succeed after many attempts have failed. This is the accepted pattern in science. Space science does not enjoy the same latitude because it is so expensive and because the public takes failures so personally. It is partially the public's self-identification with the space science program that makes them willing to finance it. Therefore, the restrictions placed upon space science become all the more fearsome when a rival nation is not only successful, but appears to have succeeded on its first attempt. Long-range flight scheduling becomes at the same time both essential and precarious. At any time it may be necessary to drop the score and play by ear.

Congressional hearings which culminated in the Space Act of 1958 make it clear that the preservation of the national reputation and the efficient pursuit of space knowledge are two of the Act's primary goals. International competition has made the American public feel more keenly about its reputation than about any other of the Space Act goals. This social situation poses the question: "Is it possible for the space laboratories to conduct an efficient research and development program with the nation peering over their shoulders and periodically demanding spectacular action?"

In attempting to assess public reaction to an impending space program, the opinions of the Central Intelligence Agency (CIA), the United States Information Agency (USIA), the National Academy of Sciences (NAS), and numerous laymen were solicited. Some of the more important questions and their answers are listed below.

1. *Question:* What space experiments intrigue people most?

   Answer: Something leading to the detection of life in some other part of the universe. This problem interests people far and away more than anything else. There is a tendency to confuse this goal with the engineering achievement of putting a man in space.

2. *Question:* How do people respond to failure in space experiments?

   Answer: It takes some of the surprise element and hence psychological impact out of a later success. The psychological payoff is not as high. However, failures are considered as worth it, if a "first" is obtained. "Firstness" is much more important than avoiding failure.

3. *Question:* What experiments will possibly offend the world public?

   Answer: Many people would be offended by radioactive contamination, and possibly by the chemical and biological contamination of other planets. This is not something that currently worries the general public. The same groups that are vitally concerned with contamination and conservation on the earth tend to be concerned. One source indicated that an attempt to put a man in space which resulted in his death would seriously disturb the public.

4. *Question:* What experiments will worry the world public?

   Answer: There is not much to be said here, except that some people have an aversion to earth reconnaissance satellites. This is part of a feeling that "Big Brother" is looking over their shoulders.

5. *Question:* Should space flights and experiments with military overtones be avoided?

   Answer: No! Most people feel that all space experiments have some military connotations. (It was intended that the American IGY and *Vanguard* programs not be associated with military programs. However, this point was not understood or accepted by the public.) A guidance-development experiment would have clear military implications, yet an American success in this field would be very well received.

6. *Question:* Cooperation by the United States with other nations is one of the goals of the National Space Act. What geographical areas are most sensitive to their being included or excluded from the Space Program? Where would the political payoff be the greatest?

   Answer: (1) India, (2) Egypt, and (3) Japan. Unfortunately, Egypt does not appear to have much to contribute to the space exploration program. The USIA tentatively suggested that tracking stations in the Philippines and in Indonesia would be helpful.

7. *Question:* What were the principal political gains made by the Soviet Union with *Sputnik I*, etc.? Did the United States lose face?

Answer: The United States did not lose prestige directly, but Russian statements gained credibility. This has been particularly noticeable in the Near East.

8. *Question:* How should advance publicity on the nature of space experiments be handled?

Answer: This question is apparently too difficult for most people to answer. The CIA emphasized that suppressing the news on firings was very difficult and, indeed, rather precarious. Distinguished scientists generally gave emotional responses to this question, indicating their displeasure with most news reporting. They appear to favor announcing firings or experiments only after the event. However, they were very critical of the one instance where a satellite firing was kept secret until after the launching.

9. *Question:* Should considerable effort be put into an educational program utilizing mass communication media, such as motion pictures and television?

Answer: The CIA and USIA emphasized that problems of properly informing the public could not be left to Madison Avenue. Scientists must take the initiative and responsibility of learning the techniques of presenting information to the public. Scientists will need the assistance of professional public relations personnel, but the principal responsibility of formulating an educational program must rest with the space scientists themselves.

10. *Question:* What are the rocket capabilities of the Soviet Union, and what are their plans?

Answer: The Soviet Union is capable of a moon miss (subsequently verified) or perhaps a hit. The USSR has been working intensely on space-flight biology, and may be able to orbit a man and return him to earth within a year.

11. *Question:* What are Russian plans for decontaminating payloads?

Answer: A search by the CIA did not turn up a single statement of fact or philosophy. G. A. Tikhov of the USSR is quoted as believing in the existence of life on Mars. However, Oparin, a Russian authority on the origin of life, is quoted as not believing life exists on the other planets.

12. *Question:* Should the results of scientific experiments aboard space probes be made available to everyone as soon as it is physically feasible to do so, or should it be released in the traditional way at the scientist's discretion?

Answer: A poll of one hundred engineers and scientists showed a 5-to-1 preference for making the results public as soon as possible.

Public Reaction Addendum

Launching of a Rocket System Toward the Moon
by the USA

Pravda, March 4, 1959, page 6

New York, March 3 (TASS)—The rocket system "Juno II," which was launched today toward the moon from the USA, consists of four stages. The final stage carries an apparatus with instrumentation, which weighs 13 lb (5.89 kg).

According to a communique by United Press International, by 5 P.M. Moscow time the rocket was at an altitude of 72,400 miles and was traveling with a velocity of 6,518 miles per hour.

As was announced by the Reuters Agency, the leader of the laboratory for reactive motors by the National Administration for Aeronautics and the Investigation of Cosmic Space, Dr. W. Pickering, has made a statement that "the rocket has gone off course" from the predetermined trajectory, but that he does not know how far off course it went.

V. SCIENTIFIC CONSIDERATIONS

A. Background

Until such time as permanent bases are established on the various bodies of the solar system, a period which, even for the moon, is some 10 to 15 years in the future, man's ability to study these solar bodies will be severely limited by the mechanical constraints discussed in preceding Sections. Three categories of study must be assigned high priority within the program of both remote and manned investigation: (1) Is there any extraterrestrial life? (2) What can be learned of the origin and evolution of the solar system and its multitude of component bodies? (3) What must be learned to assure the success of the more difficult missions of the future? The success of any program of interplanetary exploration lies in a partial answer to these questions.

The three categories named are not restrictive. Almost any data fits into one or more of them. Questions of feasibility already discussed are far more restricting. Yet, these three categories give direction to the scientific program, and they provide a framework within which the knowledge gained may be considered.

1. The origin of life. From classic Greek times until the late 19th century, it was generally accepted that living matter, in one form or another, could originate spontaneously from non-living material. The frequently observed presence of insects, worms, frogs, etc., in mud or decaying organic matter was considered proof that these animals were generated spontaneously, without parents. This conception was disproved by Redi in 1668, but it was revived almost immediately following the discovery of microorganisms by Leeuwenhoek in 1675. For technical reasons, disproof was difficult in the instance of microorganisms. Also, people clung to this belief because bacteria, so small and apparently simple, seemed to be in the twilight zone between living and non-living matter. (Actually, these organisms are as complex as any cell of our own bodies.) In 1862, spontaneous generation of microorganisms was finally disproved by Pasteur.

As a result of Pasteur's experiments, many scientists, especially physical scientists, came to the conclusion that these studies demonstrated the futility of inquiring into the origin of life. These scientists proposed that life had no origin, but, like matter, was eternal. This was the view of Arrhenius, Helmholtz, and Lord Kelvin. Arrhenius, especially, elaborated this idea which he called the theory of panspermia. He proposed that life-bearing seeds are scattered through space, and that they fall on the planets and germinate wherever conditions are favorable. In addition, Arrhenius suggested a mechanism by which spores with diameters of the order of 0.1 micron could be carried beyond the gravitational field of the planet of their origin and be propelled (by light-pressure) through space to other planets. From this theory, Arrhenius concluded that living things throughout the universe should consist of cells composed of carbon, hydrogen, oxygen, and nitrogen.

The panspermia theory is much less attractive today than it was fifty years ago. Life is now regarded as a manifestation of certain molecular combinations. Since these combinations are not eternal—indeed, neither the elements nor matter itself are eternal, according to modern cosmologists—it is impossible to accept the idea that life has always existed. In addition, the escape of spores from the gravitational field of the earth and their survival in the unfiltered radiation of outer space seem much more difficult problems to us than they did to Arrhenius.

As a result of the increasing knowledge of the chemical nature of living matter, there is a renaissance of the idea of spontaneous generation, this time at the molecular level. In the 1920's, Oparin and, independently, Haldane, proposed that the origin of life was preceded by a long evolution of organic compounds of ever-increasing complexity on the earth's surface. In the pre-biotic, sterile world, these compounds could accumulate in the seas and eventually, by random combinations, produce a living molecule or molecular combination. (Differences of opinion as to the nature of the first living thing are ignored here.) Oparin pointed out that the synthesis of organic compounds requires reducing conditions, since these compounds are unstable in the presence of oxygen. He proposed an atmosphere of methane, ammonia, water, and hydrogen for the primitive earth. Later, Urey showed that methane, water, and ammonia are the stable forms of carbon, oxygen, and nitrogen in the presence of excess hydrogen. Since hydrogen is the predominant element of the cosmos, it is reasonable to assume that it was present in large amounts on the primitive earth. Urey suggested that ultraviolet light could provide the energy for organic syntheses in the primitive atmosphere. Model experiments by Miller have shown that organic compounds, including amino acids, organic acids, and urea, are in fact produced when ultraviolet or an electric discharge is passed through such an atmosphere.

Finally, modern genetics and evolutionary theory show that it is possible, starting with a single living particle in an environment rich in organic compounds, to account for the evolution of all living species.

The discovery of life on another planet would be one of the momentous events of human history. Such a discovery, however, would do more than answer a universal curiosity; it would also be of enormous scientific interest. Next to the synthesis of living matter in the laboratory, it would be the most important step that could be made toward an understanding of the problem of the origin of life. The question of the uniqueness of systems based on nucleic acids and proteins as bearers of life is one of the fundamental questions such a discovery might solve. It is recognized, however, that some difficult biological problems are involved here. If living forms based on protein and nucleic acid are found on another planet, what will it be possible to say about the question of independent origin versus common origin by a mechanism of the Arrhenius type? Although it is unlikely that large numbers of spores could escape from the gravitational field of the earth by the electrostatic mechanism proposed by Arrhenius, the possibility of escape of an occasional spore cannot be excluded. Important information which would bear on this question can be obtained now, in the vicinity of the earth. For example, it would be desirable to learn more about the vertical distribution of microorganisms in the atmosphere; also, more information about the ultraviolet flux in space is essential in order to estimate the chances of survival of spores.

Another basic biological problem that should be considered in advance is that of recognizing living material that is not chemically similar to our own. Such organisms may have metabolic rates and growth rates much lower than anything with which we are familiar.

Although the possibility of detecting and studying life on other planets is the most exciting aspect of space exploration, the biological importance of space research is not limited to the study of extra-terrestrial life. If no life is found on other planets, the possibility of sampling the organic compounds on other worlds can yield invaluable evidence bearing on the origin of life. These sterile worlds may well provide unique clues to the organic chemical processes which preceded the development of life on the earth. The recent observations about Mars by Sinton and about the moon by Kozyrev make it appear likely that large-scale chemical processes involving carbon are taking place on these bodies. Although it is virtually certain that life occurs elsewhere in the universe, the *a priori* likelihood of its being found on other bodies of our solar system, especially the moon, is not high; consequently, the importance, for biology, of geochemical research of the type mentioned should not be minimized.

Briefly, there is reason to believe that the results of space exploration will be of biological interest regardless of whether or not extra-terrestrial life is actually found. For this reason, it is all the more important to minimize contamination, either chemical or biological, of the moon and planets. It should be possible to set up tolerance limits for contaminants and rules of procedure which will safeguard the possibilities for significant biological investigations in space without impeding the development of other programs.

2. The origin of the solar system. Many different theories of the origin of the solar system have held the attention of "thinking men" for varying lengths of time. Gradually, the theories have become progressively less metaphysical and more scientific. As knowledge of the solar system has increased, the requirements placed on proposed theories have become more stringent. As a result, the original nebular hypothesis of Kant and Laplace and the whole spectrum of encounter theories of Chamberlin and Moulton, Jeans and Jeffreys, and Lyttleton have been effectively disposed of in recent years. (Lyttleton's binary encounter theory, if not proved impossible, is at least highly unlikely.) Two major theories exist today, the "hot" theory of Alfvén and the "cold" theory of Kuiper (an extensive revision and enlargement of ideas introduced by von Weizsäcker) with geochemistry added by Urey and others. No one man is responsible for all of the details of either theory, of course, but the men named have presented these theories in the most complete detail seen in print to this time.

Both the hot and cold theories picture the existence of a solar nebula some billions of years ago. In these theories, the planets are visualized as beginning to form from the nebula during the last phases of solar formation. (The existence of appropriate conditions agrees with present cosmological theories and theories of star formation and evolution.) At this point, the hot theory assumes that the degree of ionization was sufficient to make electromagnetic forces play a considerable, if not dominant, role in planet and satellite formation. The cold theory assumes a purely dynamical process of formation in an uncharged medium until late in the formation process.

At the present time, Kuiper's cold theory enjoys the widest acceptance among astronomers. It explains in considerable detail items which Alfvén has not even discussed. This wide acceptance of Kuiper's theory may be due, in part, to the newness and inherent difficulty of magnetohydrodynamics, a field in which few astronomers feel comfortable. If the past is any indication, the final answer may contain elements of both theories.

Kuiper's theory avoids both the gravitational-instability and angular-momentum distribution problems (which destroyed Laplace's nebular hypothesis) by postulating a nebula containing between 5 and 10% of the solar mass. So massive a nebula would be gravitationally unstable and would break into discrete clouds. These massive discrete clouds of matter called protoplanets would have been stable in the solar tidal field and would have contained most of the angular momentum of the solar system. According to Kuiper, condensation took place within these protoplanets, forming planets near protoplanet centers with surrounding extremely extensive disk-shaped atmospheres of essentially solar composition. Satellite formation took place within a given protoplanet as an essentially similar repetition of the original planet formation. Satellites were relatively closer to their primaries than the planets were to the sun; therefore, tidal friction caused equality of periods of rotation and revolution. The slow rotation meant near-spherical symmetry for the satellites and further breakup into satellites of satellites did not occur. Similarly, solar tidal friction was able to prevent Mercury and Venus from rotating rapidly enough to form satellites.

During the planet formation process, the sun continued to shrink and become hotter, and, after about 10^8 years, it achieved its present equilibrium size. The intense radiation from the sun then swept away most of the huge gaseous envelopes of the protoplanets; thus, in the case of the nearby terrestrial planets, only the central condensations remained. The more distant major planets were able to hold some of their hydrogen and helium, and, as a result, have a structure differing radically from that of the terrestrial planets. Kuiper believes that Pluto was once a moon of Neptune rather than a planet in its own right, which would explain its differing from the other outer planets.

It is believed that the asteroids have formed in a region in which the density of the solar nebula was too low to form one protoplanet. Subsequent collisions greatly increased their number. (See Sec. V-F-1 on asteroids.)

Comets are composed primarily of frozen gases, and it is believed that they have formed far out in the solar system, beyond Pluto, out of the gases which formerly made up the protoplanet envelopes.

Most of the dynamical features of the solar system as we know it today can be explained by Kuiper's theory (which has been worked out in some quantitative detail, as well as in far more qualitative detail than has been given here). Immediately following the appearance of Kuiper's dynamics, Urey and others undertook study of the geochemical problems involved in the theory. Two major problems appeared to exist. The first was the fractionation of silicate and metallic constituents in the terrestrial planets. The second was the lack of fractionation (and dissipation) of the more volatile elements of meteorites in spite of the obvious high-temperature melting of silicates and metals therein at some time in the past.

Urey has not been satisfied that minor modifications of Kuiper's theory will completely explain the geochemical problems. As a result, he has presented the following highly modified theory: "It is suggested that objects of considerable size and mass accumulated in a dust cloud at low temperatures. Intermittent chemical heating may have produced molten silicate and metal phases. After some period of time, millions or tens of millions of years, these objects fell rapidly toward a gravitation center, and the Sun was formed with a disk of residual gas and these objects. Melting of the silicate and metallic materials may have occurred during this process. In this process, intense collisions occurred which reduced the solid material to fragments varying from micron sizes and smaller to the sizes of the materials of the chondritic meteorites and iron meteorites. The gas and dust were dissipated from the disk by solar light pressure and turbulent gases; some of the solid material was removed by the Poynting-Robertson effect. The planets and asteroids accumulated during this time. Protoplanets in the sense of masses of gas and solids of solar composition did not occur, at least in the case of the terrestrial and minor planets, or, if protoplanets were present at all, they did not form until the permanent gases were removed and the fractionation of the silicate and metallic phases was complete, and hence these protoplanets did not have the mean solar composition. The satellite systems of the major planets were formed by analogous processes."[6]

Shortly after publication of the paper cited above, Urey realized that the primary objects were exactly like the moon, and he formed the opinion that the moon is a primary object and that it was not broken up and captured by the earth during its growth.

These ideas are all highly controversial, of course. The dynamicists are brought to task for ignoring geochemical arguments, and the geochemists are in turn chastised for impossible dynamical arguments, for this "astro-archaeological" research, of necessity, draws from all fields of physical science in a manner common to few other disciplines. Nevertheless, this represents the type of thinking which must be applied to our study of the solar system if the results are to be something other than a collection of disjointed, apparently unrelated facts.

Following the period of origin, which occurred between 4 and 5 billion years ago according to the best estimates, the solar system began to evolve. In the case of Venus, earth, and Mars, all exterior evidence of the days of formation disappeared through eons of geological activity. However, one "easily" accessible body offers possible information on this early period. That body is our own satellite.

B. The Moon

The nearest, and in one sense perhaps the most interesting, body in the solar system is the earth's own satellite, the moon (see Fig. 55). The moon is a prehistoric remnant, a body relatively unchanged, according to most present day theories, for more than 4 billion years. It can be hoped, with some reason, that the moon will prove to be the Rosetta Stone that unlocks many of the secrets of the origin and evolution of the solar system.

Astronomers already possess considerable knowledge of the moon. Some of the best specialists in the history of celestial mechanics investigated the problem of the moon's motion, and from a theoretical standpoint, the culminating treatise of E. W. Brown is more than adequate today. However, some of the constants of the theory, such as the mass of the moon, could easily be in error by as much as 0.05%.

Pertinent Physical Data

Mass	0.012304 of earth
Radius	0.2728 of earth, 3476 km
Mean density	3.34 g/cc
Axial rotation	$27^d\ 7^h\ 43^m\ 11.5^s$ mean solar
Mean surface gravity	0.165 of earth
Albedo	0.07
Velocity of escape	2.24 km/sec
Apparent angular diameter (mean equatorial)	31′ 05″
Mean distance	238,857 mi, 384,403 km
Orbital eccentricity	0.0549
Sidereal period	$27^d\ 7^h\ 43^m\ 11.5^s$ mean solar
Temperature range (maximum measured)	134°C to −153°C

The moon's periods of rotation and revolution are identical; hence, the same face is always seen. However, because of geometrical and physical librations, about 59% of the surface can actually be seen at one time or another. This value of 59% has been very thoroughly mapped in two dimensions, as can be seen from the most recently published selenography (geography of the moon), that of Wilkins and Moore.[7] All detail smaller than about one kilometer is necessarily missing since this quantity is about the limit of resolution imposed by the atmosphere. Vertical dimensions are available at only a limited number of points since only a few hundred are accurately determined.

Studies by Gilbert, Baldwin, and others have made it very plain that the great craters on the moon are impact craters rather than volcanic craters (see Fig. 56). It has also become clear that at least some of the maria (the great plains) are the direct result of impact (see Fig. 57). It is not clear if the impacts acted primarily as a trigger mechanism releasing molten material from the moon's interior, as proposed by Kuiper, or if the molten material resulted primarily from the kinetic energy of the impacts,

[6]Urey, H. C., *Astrophysical Journal*, 124:623, 1956.

[7]Wilkins, H. P., and Moore, P., *The Moon*, Macmillan Co., New York, 1955.

Fig. 55. Composite Photograph of the "Full" Moon

Harold C. Urey

Fig. 56. Region of the Lunar Crater Clavius (Photographed Through the 200-in. Telescope)

Mt. Wilson and Palomar Observatories

as advocated by Baldwin and Urey. The argument, in part, revolves about whether or not the moon was ever in large part molten.

The moon is an ellipsoid with its greatest axis pointed toward the earth. The least axis is essentially normal to the plane of its orbit. The long axis (the lunar tidal bulge) is "7200 $\pm$ 600 ft greater than the average diameter in the plane of the sky" according to Baldwin's reduction of Franz and Saunder's direct observational measurements. Dynamical calculations give a similar (6280 ft) and completely independent result. On the other hand, if the moon were free to completely adjust to the earth's tidal pull and its own centrifugal forces, the total diametric bulge should be only 376 ft.

Two explanations have been proposed for this difference: There is definite dynamical evidence that the moon was once much closer to the earth than it is today. Tidal friction has caused an exchange of rotational angular momentum (of the earth) for orbital angular momentum (of the moon) which shows up primarily in an increase in the moon's mean distance. If the moon solidified when at a distance from earth of only 100,000 km (as it was about 4.3-billion years ago), the presently observed bulge is simply the "frozen-in" gravitational equipotential bulge of that long-ago time. This would imply that the moon was never molten during the ensuing 4.3-billion-year period (an implication strongly advocated by Urey).

The other possibility, proposed by Kuiper, is that of preferential accretion through an encounter with a so-called sediment ring. In the modern von Weizsäcker-Kuiper nebular theory, a sediment ring is a ring-shaped swarm of relatively small bodies which forms "where the density of the nebula is insufficient to cause breakup by gravitational instability." The ring of Saturn and the asteroid belt are two possible examples. It is assumed then that, as the moon moved outward, it encountered a sediment ring. The moon's rotational period is assumed to have already become coincident with its period of revolution, therefore preferential accretion took place. Later, involved dynamical arguments indicate the possibility of motions which would cause the long (added to) axis to become first the axis of rotation and then the axis pointed toward the earth. It is obviously imperative that the moon be relatively rigid by this time, or the bulge would simply isostatically adjust and disappear. It must be noted that the dynamical arguments involved in this procedure have never been worked out in detail.

Kuiper argues that the impacting debris released molten material from inside the moon, forming the maria and generally reshaping the moon's surface. One great difficulty of this theory is that the moon must have been rigid enough to support a bulge, yet molten in enough regions to supply lava in great quantity.

Although the great craters are meteoric in origin, this does not imply that no volcanic activity can exist on the moon. On the contrary, there are rows of craterlets near Copernicus (one of the great craters) which are almost certainly due to volcanic activity. One of the most interesting discoveries in the past few months was by the Russian astronomer Kozyrev. In a portion of the meteoric crater Alphonsus, a temporary haziness was found which lasted long enough to obtain a spectrogram confirming the existence of C_2, C_3, and other as yet unidentified species. Thus, gases do exist, at least for a short time, on the surface of the moon. This "atmosphere" is very tenuous at best ($<<10^{-7}$ atm) and must consist mostly of the

Fig. 57. Looking North from Copernicus Across Mare Imbrium (Photographed Through the 100-in. Telescope)

Mt. Wilson and Palomar Observatories

heavy inert gases such as xenon except immediately after a blow-off such as the one in Alphonsus.

It has been suggested by Gold that the maria consist of deep layers of dust produced by ultraviolet radiation from the sun and by alternate heating and cooling of the moon's surface. This cannot account for some of the more obvious collision features such as Mare Imbrium but could account for some observed features. In many situations, Urey accepts the theory of dust as a useful explanation, but he favors a theory of collisions as the source rather than Gold's mechanism. Radio measurements have indicated that there is indeed dust to a depth of at least a millimeter over some large fraction of the moon.

Having considered some of the known and postulated facts about the moon, how may our ideas best be confirmed or denied, and our knowledge increased? First, measurements of the moon's uranium, potassium and thorium abundances (by means of a γ-ray spectrograph), thermal conductivity, and temperature (far enough below the surface to escape rotational effects) will tell much of the thermal history of the moon. Seismology will tell of the moon's internal structure (thought to be homogeneous), moonquakes, and tidal effects caused by the earth. Gamma-ray spectroscopy, X-ray fluorescence, or eventually actual chemical analysis will tell us if the maria and craters are granite or basalt, containing lumps of iron-nickel, or perhaps undifferentiated silicates and metals. This will tell us whether or not the moon is indeed a "primary" body and whether the lavas came mostly from the interior of the moon or just from surface melting. Local magnetic-field measurements will give information on the local abundance and distribution of magnetic materials; whereas, mapping of the general field (if any) will have strong implications concerning the history and constitution of the interior, though interpretation will be difficult (there being no real agreement on the theory of the earth's field). An orbiting vehicle, if properly tracked, will give new mass and mass distribution figures.

One of the most useful of all possible lunar missions will be that of photography. Much of what is known about the moon was learned by careful study of the geometric relationship of its various features. Closeup mapping of the known side and almost any information on the unknown side, the side never seen from earth, cannot fail to be useful. Photometric measurements of albedo and polarization as a function of wavelength could give useful indications of surface composition. A mass spectrograph could analyze the lunar atmosphere.

The moon is an airless, waterless dead body which alternately "freezes" and "boils." The existence there of life as we know it is unthinkable. Still the possibility of the existence of so-called sub-life forms must be considered. The action of atoms and molecules at the surface of the moon under eon-long bombardment by undiluted solar radiation and by cosmic rays cannot be predicted. The formation of complex macro-molecules may be possible or even probable. Instrumentation for the detection of such species must be considered. How to perform this detection, without at the same time negating the detection through contamination, is an unsolved problem as yet.

In order to ensure the success of later expeditions to the moon, a few precautions must be taken: (1) The moon must be checked for a radiation belt. Such data are necessary to ensure successful use of photomultipliers and other radiation-sensitive equipment in the vicinity of the moon. (2) It would be useful to know something of the surface harness of the moon in various areas to help guarantee a safe first soft landing. (3) It would be useful to have detailed photographs of at least one mare surface to aid in the design of landing gear appropriate to that surface. (4) Environmental testing of varieties of equipment under the extremes of lunar conditions will be necessary either under simulated conditions here on earth or actually on the moon.

C. Venus

Considering that Venus can approach within 26 million miles (42×10^6 km) of earth, nearer than any other planet, it is discouraging that almost every other factor is against the successful attainment of any considerable knowledge of the planet. When Venus is nearest earth, at inferior conjunction, it is almost unilluminated since it is between the earth and the sun. At greatest elongation (greatest angular separation from the sun), Venus is about 64 million miles (103×10^6 km) distant, which is almost twice the distance of Mars at its most favorable opposition. At superior conjunction, the only time when the disk is completely illuminated as seen from earth, Venus is about 160 million miles (257×10^6 km) distant, presents a disk of less than 10-in. angular diameter, and is always less than 2 deg from the sun. In addition, Venus is covered by very opaque clouds which completely hide its surface (see Fig. 58). Finally, Venus has no satellite; therefore, even the mass is uncertain to "a considerable fraction of 1%." The importance of accurate masses should not be underestimated. Geological and atmospheric structure studies, planetary celestial mechanics, and probe trajectories are all strongly dependent upon accurate planetary masses.

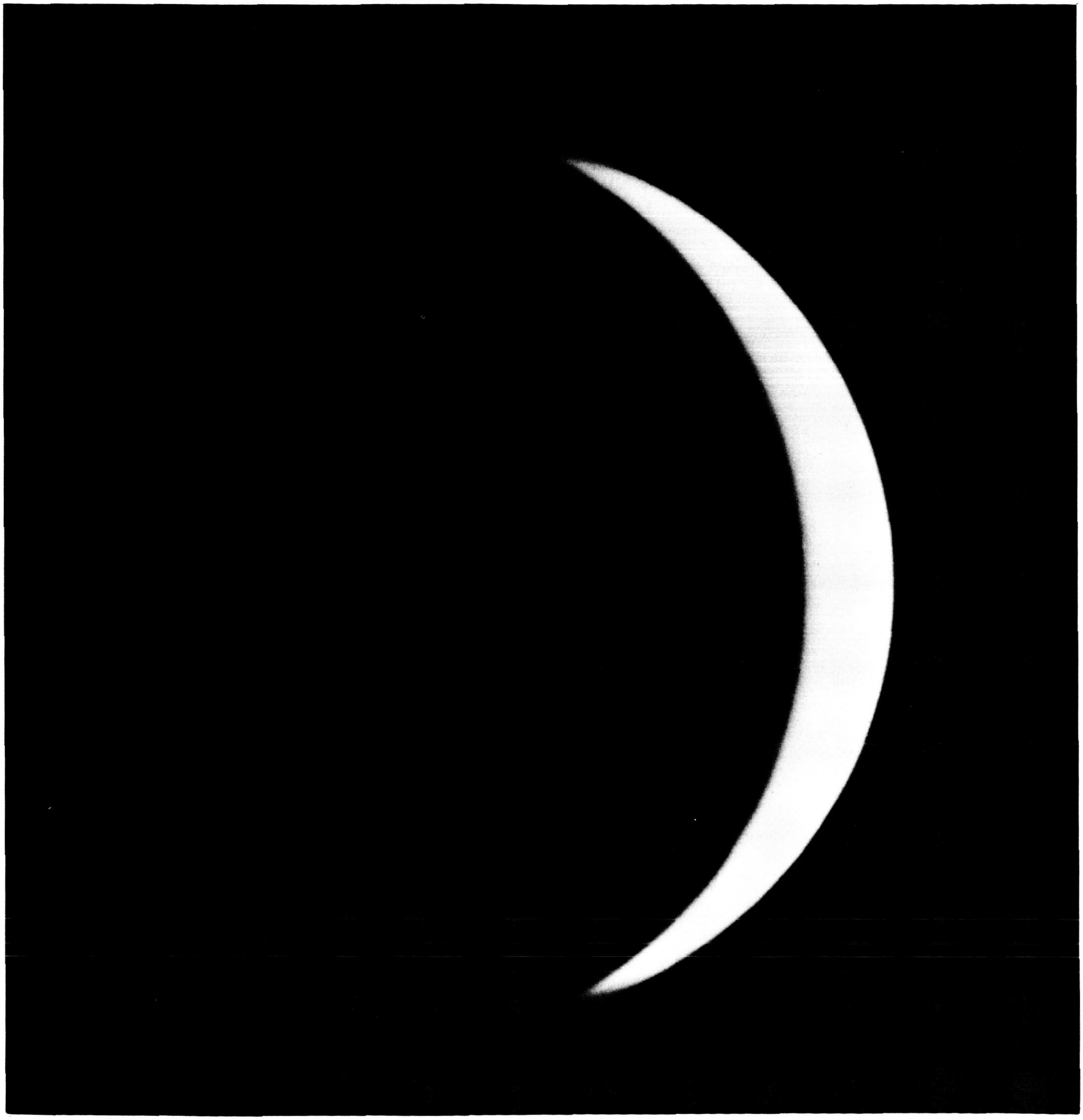


Fig. 58. Crescent of Venus (Photographed in Blue Light Through the 200-in. Telescope)

Mt. Wilson and Palomar Observatories

What then is known about Venus? Its orbital elements have been known accurately for many years. Venus revolves about the sun in an almost circular orbit (eccentricity, only 0.00681) at a mean distance of 0.723331 A.U. in a plane inclined 3° 23′ 38″ to the ecliptic. Its sidereal period is 224.7008 mean solar days.

Other measured planetary constants include Rabe's mass for Venus determined from perturbations on Eros as 0.8136 of earth. The mean diameter (actually that of the clouds, of course, rather than the solid surface) is 0.973 of earth, with a probable error between 1 and 2%. This implies a mean density of 4.87 and a mean surface gravity of 0.86 of earth. The albedo usually quoted is 0.59. A recent determination by Kuiper gives 0.68 over the entire electromagnetic spectrum. The oblateness is too small to be measured.

It is unknown whether Venus rotates on its axis or if it always presents the same side toward the sun. Temperature measurements on the day and night sides differ but little, which seems to imply a rotation. However, wind currents on a nonrotating body might produce such an equalization. Persistent attempts to obtain a spectroscopic period have been unsuccessful, the doppler shifts indicating little or no radial motion with respect to earthly telescopes. Such a result implies either a rotational period longer than about 4 days or an axis inclined toward the earth at the time of the observations. The period of rotation was recently estimated by Kraus at $22^h\ 17^m$ on the basis of modulation of an 11-meter radio source which is apparently present on Venus, but others have been unable to reproduce these results. In summation, it may be said that if Venus rotates, no one knows at what speed, at what axial inclination, or whether the rotation is direct or retrograde.

With respect to temperatures, the situation is improved. Recent measurements in the infrared (10-13 micron) have indicated a temperature of about −38°C above the cloud surface, a point high up in the Venusian atmosphere. Infrared measurements at 8 microns indicate that the temperature is 16 to 27°C within the clouds. Radio measurements at 3.15 cm and 9 cm have indicated a temperature of 300°C ± 100°, probably at the surface of the planet. The earth moved to the distance of Venus from the sun (ignoring changes in atmosphere, etc) would rise in temperature to a new mean temperature of about 69°C.

The Venusian atmosphere not only hides the surface but is a puzzle in itself. The one substance definitely known to be present in the atmosphere is carbon dioxide. The absorption features indicate a strength equivalent to 1000 meters of the gas at standard temperature and pressure; that is about 500 times the earthly amount. Oxygen and water vapor have never been observed, and, if present at all in the part of the atmosphere that can be studied spectroscopically, the quantities of these substances must be less than 1% of that in the earth's atmosphere. An emission feature has been seen in the spectrum of the dark side at the frequency of known bands of the singly ionized nitrogen molecule. This could be an effect of the earth's atmosphere or even some other molecule, but the implication is strong that there is nitrogen on Venus.

The composition of the clouds is an important consideration. The once popular formaldehyde hypothesis has fallen into disrepute, and at present there are three prominent theories. These are: the clouds are dust, polymers of carbon suboxide, or water in the form of liquid droplets or ice crystals. The clouds actually seem much too white to be dust. Dust would seemingly cause reddening of light scattered from it unless an unusual spectrum of particle size and composition is involved. The idea of carbon suboxide polymers is only speculation since it isn't known that such polymers are even possible. Such ideas have arisen because of the known large concentration of carbon dioxide. If −38°C is actually the correct temperature at the cloud surface, any water would be present as ice crystals and, therefore, undetectable spectroscopically.

One other fact has been observed: Markings are visible on ultraviolet photographs of Venus. These markings change from day to day, however, and are apparently part of the cloud structure.

As to what is underneath the clouds, speculation makes Venus everything from a boiling wind-blown desert to completely ocean covered. Drawing inferences from what earth might have been like had it been placed in Venus' orbit, Dole suggests a dry planet with a surface atmospheric pressure of 8 to 10 atm consisting of 90% by weight carbon dioxide and most of the balance nitrogen. In fact, nothing is known about the surface of Venus.

In probe study of Venus, the first step must include magnetic-field and cosmic-ray trapping equipments to check on the instrumental environment for later Venus shots. In the event that the probe does not get near enough to Venus to sample the radiation belt (if any), the equipment on board still will operate to sample the general interplanetary medium; thus, nothing is wasted. Equipment should also be on board for high-resolution photographs at several wavelengths from ultraviolet to infrared, the purpose being detailed study of the cloud structure.

It is probable that by this time spectrograms of Venus in the vacuum ultraviolet will have been taken from earth satellites. These spectrograms or, perhaps more practically, these ionization-chamber measurements will be investigated with special reference to nitrogen and argon abundances. If such is not the case, such measurements must be made from an early probe.

The first Venus satellite will necessarily be instrumented to continue atmospheric studies. A microwave spectrograph should be flown to allow penetration below the clouds and even clear to the surface. Suggested operating frequencies are 3 cm, 1.35 cm, and 0.5 cm to allow a determination of oxygen and water vapor in the lower atmosphere. Such equipment will also give a low-resolution mapping of the surface and, according to Gold, might indicate the presence of life forms through the resulting roughness measure. Measures in the far infrared with photoconductive devices would supplement the microwave measurements. Wide-band infrared photography as well as narrow-band spectroscopy would seem to be in order. A second magnetometer and cosmic-ray package should also be carried.

The primary mission of the first Venus entry shot will be to pave the way for the first soft landing. As this package penetrates the atmosphere, a complete temperature, pressure, density profile must be obtained. In addition, a beacon transponder should be planted on the surface of Venus with the added ability to determine whether the surface upon which it has landed is liquid or solid and, incidentally, the nature of the atmospheric transmission characteristics. If additional weight and bandwidth are available, a mass spectrograph and/or an infrared camera should be carried.

The first soft-landed package should be a sophisticated multipurpose device carrying a complete weather station which includes sounding balloons. The package must have rapid high-resolution picture-taking ability in all directions and at varying distances. It should provide for chemical and microscopic analysis of the surface upon which it has landed as well as a few feet below the surface. It should include a listening device sensitive to a wide acoustical spectrum. Provision should be made for detection of 3.4- to 3.5-micron absorption due to the CH band as a possible indication of life. Seismic and magnetic-field measurements will also be in order if the weight and bandwidth are available.

At the successful conclusion of such a series of experiments, a considerable amount of knowledge of the geological history and present state of Venus will have been gained. In addition, any indication of life in the area of the vehicle will have been noted, as well as information as to the adaptability of Venus to human life (or vice versa).

D. Mars

Mars has been the subject of more study and speculation than any body in the solar system except the sun. It comes closer to earth than any planet except Venus [to within 34.5 million miles (55.5×10^6 km)] at a favorable opposition, and unlike Venus, it is completely illuminated at the time of closest approach. Although Mars has an atmosphere with clouds, surface markings can still be seen and add to the interest (see Fig. 59). In fact, the knowledge already possessed has only made it seem more important to obtain additional information.

Pertinent Physical Data

Semimajor axis	1.523688 A.U.
Eccentricity	0.09333
Orbital inclination	1°51′01″ to ecliptic
Mass	0.1080 of earth
Diameter (mean)	0.520 of earth, 6,620 km
Oblateness	1/192 (dynamical)
Mean density	4.2 g/cm^3
Mean surface gravity	0.40 of earth
Velocity of escape	6.4 km/sec
Sidereal period	686.9797 mean solar days
Rotational period	$24^h\,37^m\,22.58^s$ mean solar
Axial inclination	25°12′ to pole of ecliptic
Apparent angular diameter	3.5″ to 25.1″ (equatorial)

Measurement of the diameter of Mars is not particularly easy. The apparent diameter is a function of the wavelength of light in which the measurement is made, this effect being due to the atmosphere. There is a problem beyond the apparent problem, however. The mean value of the optical flattening from many different determina-

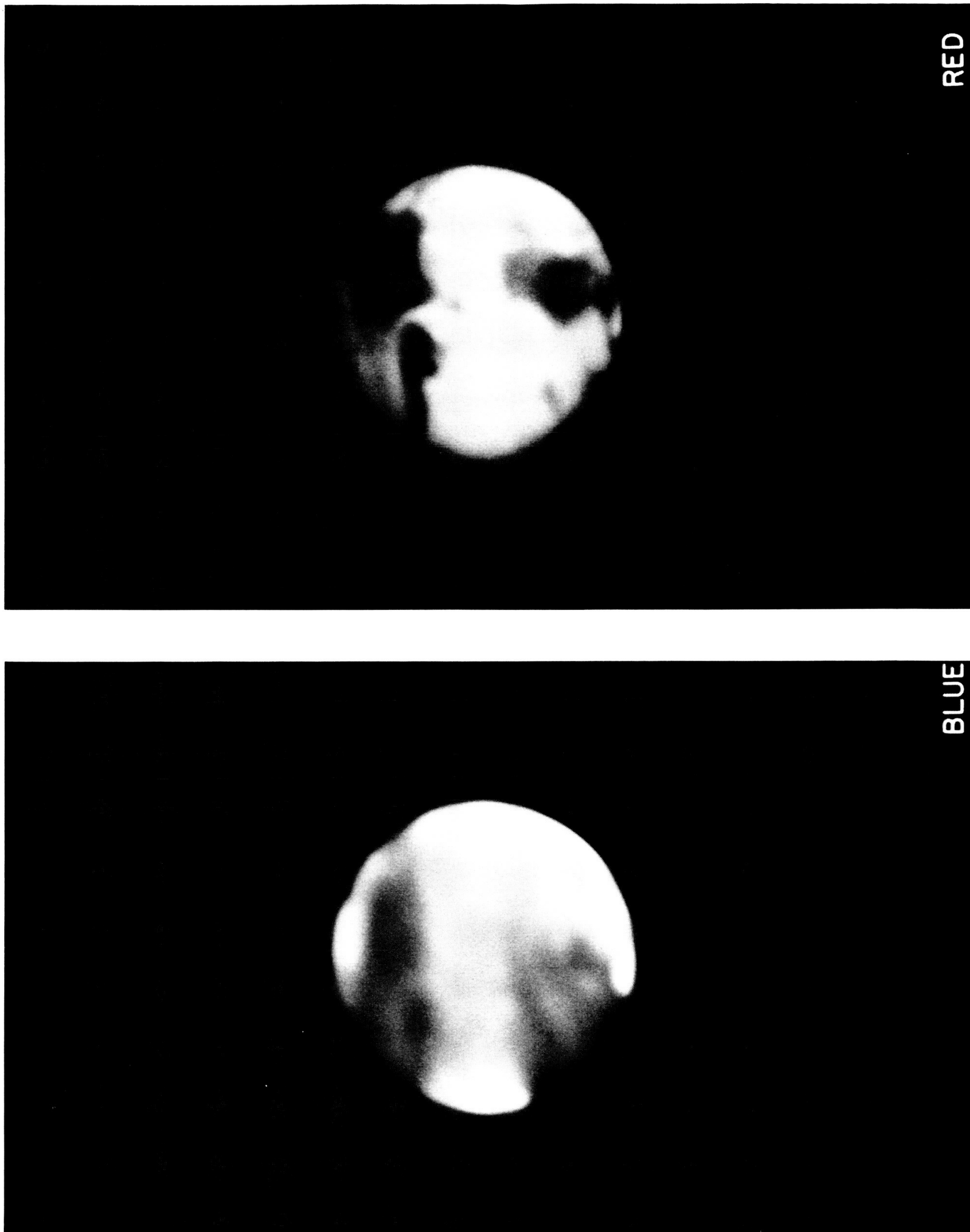


Fig. 59. Mars, Showing Atmosphere (Photographed in Blue Light) and Surface Features (Photographed in Red Light, Through the 200-in. Telescope)

Mt. Wilson and Palomar Observatories

tions in four fundamentally different ways is 0.013 ±0.001. The dynamical value of the flattening derived from the regression of the nodes of the satellite orbits is only 0.005. Although several theories have been put forth to explain this phenomenon, no one is entirely satisfactory, and this remains a problem which needs investigation. Meanwhile, structural calculations are made using the (presumably) safe dynamical value.

The atmosphere of Mars is slightly less of a puzzle than that of Venus. Again, however, carbon dioxide is the only directly positively identified substance present. Calculations from spectroscopic data indicate about 10 times as much of the gas on Mars as on the earth, or equivalently, a layer about 22 meters thick at standard temperature and pressure. (Kuiper's original reduction indicated only about 4.4 meters but did not take telluric line effects completely into account.) Neither oxygen nor water vapor has ever been positively identified spectroscopically. The upper limit upon the amount of oxygen present is about 0.15% of the amount on earth over equal areas. The upper limit upon the amount of water vapor is about 1%.

One other piece of evidence on the atmospheric composition is presented by the polar caps. Vapor pressures calculated for carbon dioxide under the measured conditions at the poles indicate that the poles just cannot be carbon dioxide covered. Recent infrared spectrometer measurements on the caps by Kuiper show the same decrease in reflectivity at 1.5 microns as exhibited by snow in the laboratory. The caps are almost certainly water in the form of snow or frost.

A large number of determinations of the surface atmospheric pressure on Mars have given a value of 64 ±3 mm or about 0.1 earth atmospheres. This result is probably not as good as the probable error would indicate but is certainly not off by more than a factor or two. Taking the measured value for the carbon dioxide partial pressure, assuming about 1.2% by volume argon by analogy with earth, and taking the remainder as nitrogen, results in a scale height of 40 km for the Martian atmosphere. Collecting these results shows:

Gases	*Thickness*, meters (S.T.P.)	*Volume*, %
Nitrogen (N_2)	1800	97.5
Oxygen (O_2)	2	0.1
Argon (A)	22?	1.2?
Carbon dioxide (CO_2)	22	1.25

Above about 30 km, the Martian atmosphere is denser than that of earth, which is an interesting effect of the lower surface gravity. Also, though the surface pressure is 1/10 that on earth, the mass of atmosphere over a unit area is between 1/5 and 1/3 that on earth.

Just as in the case of Venus, clouds present one of the puzzles of the Martian atmosphere. Three types of clouds are observed on Mars—white clouds, yellow clouds, and blue clouds. The blue type presents the greatest problem. The white clouds occur at elevations as great as 20 to 25 km, and polarization measurements seem to indicate that they are composed of tiny ice crystals similar to our own cirrus clouds. Sometimes a whiteness is observed at the rising limb of Mars, probably a fog or mist or hoar-frost, which soon disappears. The white clouds themselves seldom, if ever, persist until noon except in the polar regions.

The yellow clouds, visible in yellow and red light but not blue or violet light, are found at elevations up to about 5 km, and the indication is strong that they are dust clouds. Yellow clouds are seen more frequently when Mars is near perihelion and the surface temperature is higher. This increase is quite probably due to the enhanced convection currents at such times.

The blue clouds and the violet layer are the most puzzling. These features are prominent in blue or violet light but become invisible in the yellow or red. It seems probable that the blue clouds are just regions of greater density in the violet layer. The transmission properties of this layer indicate that the dimensions of the particles making up the layer are in the range 0.1 to 1.0 micron. If they were smaller than 0.1 micron, they would scatter according to Rayleigh's law, and if they were larger than 1.0 micron they would scatter almost uniformly throughout the visible and form a white layer.

The big puzzle is furnished by the fact that, near opposition, the violet layer often dissipates in a very short time and, therefore, the atmosphere becomes almost as transparent in the blue as in the red. This is the famous "blue clearing" phenomenon. Actually this phenomenon has been observed (at least once) nearly 3 months before opposition. The suggestion has been made that the association with opposition is due only to the fact that the planet is rarely observed except near opposition. Yet, statistics seem to indicate that there is a real association. If the composition of the violet layer were known, solution to the problem would be more obvious.

The violet layer is thought to lie very high in the atmosphere at perhaps 30 or 35 km. Either water or carbon dioxide could exist as small solid particles under the conditions at that altitude. Either supposition has its group of advocates. It has also been suggested that the layer is made up of meteoric dust of the appropriate size, but it is difficult to see just how or why such material could dissipate in the very short time in which the blue clearing takes place. This indicates that there is still a great deal to be learned about the Martian atmosphere.

Radiometric determinations of temperature at various places on Mars during various times of day and times of the Martian year have given a fairly clear picture of this important parameter. At the sub-solar point when Mars is near perihelion the temperature may reach 30°C. Near aphelion, the corresponding temperature is about 0°C. At the summer hemisphere pole, the temperature at mid-day is somewhat below 0°C, even at perihelion. The greatest latitude within reach in the winter hemisphere shows a temperature of about −60°C at that time. At night, temperatures may fall as low as −100°C even near the equator, but these temperatures cannot be adequately measured because relatively little is seen of the unilluminated portion of the disk.

The surface markings of Mars have undoubtedly caused more controversy during the last 100 years than anything else in the solar system (see Fig. 59). The nature of the polar caps has been fairly conclusively decided. They are just areas of "frost" which extend in the winter and melt back in the summer. Most of the disappearance is probably sublimation rather than true melting, but a dark fringe around the polar cap in the spring is an indication that some melting may take place.

The so-called bright areas are rather uniform orange expanses which give Mars its characteristic color. They are areas of moderate relief, for no mountains have ever been seen on the edge of the disk, and no shadows have ever been noted when Mars was near quadrature and under oblique illumination. Certainly no mountain can exceed 6000 to 9000 ft in elevation. The bright areas are usually thought to be sandy deserts, their albedo being similar to those of our own deserts. The yellow clouds, thought to be sand and dust, are consistent with this interpretation. Dollfus found that the polarization curve of Mars almost exactly duplicates that for limonite, a mineral of almost pure ferrous oxide. Kuiper, however, found much better infrared reflectivity agreement with felsite, an igneous rock made up of a silicate of aluminum and potassium with quartz and other inclusions. Coblentz has found some evidence of a silicate emission band between 8 and 10 microns in the spectrum of Mars. The desert hypothesis seems well founded, but the nature of the material making up the deserts is not definitely known.

A large number of permanent dark markings are also prominent on the surface of Mars. In the early spring, a wave of coloring sweeps over these and they are variously described as grey, bluish, greenish, or sometimes dull blue-green. As summer approaches, a new coloring wave sweeps down from the pole and these areas turn brown (some of them even violet and crimson). There is strong evidence that these areas are actually covered with vegetation of some kind. If these areas did not renew themselves somehow (through growth), it is difficult to see why they wouldn't soon become sand-covered and yellow orange like the rest of Mars. Although the actual flow of liquid water can't take place on Mars, a wave of humidity could (and almost certainly does) flow down from the poles as they sublime in the spring. This wave would be expected to cause a greening up of some sort of plant life; whereas, other explanations of the color change are hard to imagine.

The most prominent alternative hypothesis has been an alternate humidification and desiccation of hygroscopic minerals showing attendant color changes. No mineral or group of minerals is known, however, which would respond in such a striking fashion to such a small amount of humidity change, and such a widespread occurrence of such substances is hard to imagine even if they do exist. Also, the resistance to being covered by sand cannot be explained by such a process. A recently proposed volcanic hypothesis by McLaughlin seems even less tenable.

Spectroscopic observations of Mars do not show the infrared increase in reflectivity exhibited by chlorophyll. Mosses, algae, and fungi show no such feature; however, this is not to suggest that the dark areas are one or more of these but only that all vegetation does not have the 0.7 to 1.4 micron increase in reflectivity. The most suggestive argument in favor of the vegetation hypothesis has been furnished by Sinton. The fundamental vibration absorption band of the carbon–hydrogen bond occurs at 3.45 microns. Sinton has found this feature in the reflection spectrum of the dark areas; whereas, it is absent from the bright areas.

It should be noted that, although the dark markings are called permanent, changes are sometimes observed. In 1954, a large new greenish area of about 580,000 sq mi became prominent enough to attract attention. There is evidence that it began to develop as early as 1946 but only became conspicuous enough to cause a great deal of comment with the 1954 opposition.

One other feature must be mentioned, the famous "canals." The Italian astronomer Schiaparelli first noted some rather narrow linear continuous markings on Mars in 1877. He called these canali. This should have been translated streaks or channels, but was translated canals, and canals they have been ever since. Lowell and his co-workers proceeded to find several hundred of these canals including many double canals. Intersection of as many as 14 canals in one small spot was discovered. None of this can be photographed, however, due to the blurring effect of the earth's atmosphere upon time exposures. Visual observations with larger telescopes have done little to confirm Lowell's observations. There are definitely some linear markings on Mars. According to Atoniadi and Dollfus, when viewed under good seeing conditions with large aperture these markings tend to break up into a series of small irregular spots. Whether the very narrow linear markings observed by Lowell and many others are real or optical illusions remains to be seen. In any event, today most people doubt that these are anything other than natural phenomena, assuming they exist at all.

There are problems associated with the study of the internal structure of Mars as well as the surface. The classical theory of Radau and Darwin predicts the flattening of a planet in hydrostatic equilibrium in terms of observable parameters including the oblateness, and the moment of inertia. The oblateness (or flattening) can be obtained from the motion of a satellite; therefore, in effect the moment of inertia is calculated. When this process is applied to Mars, something very near to a homogeneous planet is obtained. On the other hand, Jeffreys, Bullen, and others, have calculated models of the internal structure of Mars which strongly imply an iron-nickel core of about 1000-km radius. Unfortunately, a fairly large change in internal structure at the center changes the implied flattening only slightly. More work is obviously needed. A seismograph on Mars should solve the problem, if it is not solved sooner by other means.

Mars has two satellites, Phobos and Deimos. Phobos is the only known case of a satellite with a period of revolution around its primary which is less than the period of rotation of the primary. Thus, Phobos rises in the west and sets in the east. More factually, it is 5820 miles (9365 km) from the center of Mars and has a period of $7^h\,39^m$. Deimos is 14,600 miles (23,490 km) from the planet's center and has a period of $30^h\,18^m$. Phobos and Deimos are about 16 km and 8 km in diameter, respectively, as estimated from their brightness. From the surface of Mars, Phobos would have ⅓ the diameter and about 1/25 the brilliance of our own moon as seen from earth; whereas, Deimos would appear about like the planet Venus, being only 80 in. in apparent diameter and 1/120 as bright as our moon. These satellites may one day prove very useful as space stations.

The first probes sent in the direction of Mars will necessarily contain the usual magnetometer and cosmic-ray equipment. It is necessary to get the field and particle environment of any new area of the solar system if future experiments are to be properly planned. Additionally, something about Mars itself can be learned with this equipment in the fortunate happenstance of a close approach. The other item of experimental equipment to be carried in this early probe should be photographic equipment sensitive to the red, if only one frequency is available, and to the blue, yellow, and infrared if additional filters and/or detectors can be carried. Such equipment could solve the "canal" problem permanently, should solve the optical versus dynamical flattening problem, and, if high-resolution small-area pictures are taken, could tell us something of both the biology and geology of Mars.

More sophisticated later probes should first of all do more and better photography and add to it good high-resolution spectroscopy. Particular attention should be paid to high dispersion spectra in the vicinity of the CH band at 3.45 microns. If the probe can actually penetrate the fringes of the Martian atmosphere, a mass spectrometer should be carried as well as temperature- and pressure-measuring devices.

Still later, more sophisticated soft-landing vehicles will include a complete weather station, seismograph, and chemical-analysis equipment (possibly X-ray fluorescence, possibly chemical analytical, maybe both) just as was included in the Venus soft landing. Attempts to radiodate the Martian sands should be carried out. In addition, biological experiments to study the Martian vegetation will be of primary importance. This may well imply 2 shots, one into the dark areas and one into the bright areas. If this is the case, the dark area shot should come first, the biological problem being by far the more important.

E. Other Planets

The "Other Planets" include some unusual planets. They are "other" only in the sense of being not quite as easily accessible and not quite as interesting as Venus and Mars, the two planets nearest the earth. These other planets include Mercury, a terrestrial-type planet; Jupiter, Saturn, Uranus, and Neptune, the so-called major planets; and Pluto, which no one quite knows what to call.

1. Mercury. In spite of its relative nearness, Mercury is a little-known planet. It is just too small and too near the sun. Its maximum apparent angular diameter is only 12.7 sec; whereas, the minimum is 4.7 sec. The semi-major axis of its orbit is only 0.387099 A.U. Mercury has a diameter of 4990 km (to about $\pm 5\%$), and the greatest angular distance from the sun it ever achieves (at the most favorable elongations) is 28 deg. Mercury has no satellites and is too small and distant to have much perturbing effect on other planets. As a result, its mass has had to be determined from perturbations on the asteroid Eros and is known to only about $\pm 0.5\%$. The numerical value is about 0.0543 of earth. The density is between 4.5 and 5.1, depending upon the radius used in the calculation.

One of the more important jobs for a Mercurian satellite would be to tie down the fundamental physical parameters noted in the last paragraph. Actually, it would be desirable to have all three "diameters" of Mercury, which is presumably an ellipsoid with its greatest axis pointed toward the sun. Mercury's periods of rotation and revolution are identical, just as are the moon's and for the same reason. The sun presumably raised a tidal bulge on Mercury and then exerted a torque on it. These diameters and the mean density would serve as a basis for investigation of Mercurian geo(?)physics.

Mercury possesses no detectable atmosphere. A gamma-ray spectrograph in a satellite could therefore be utilized to study the uranium, potassium, thorium ratio in the crust of Mercury just as in the case of the moon (assuming no radiation-belt interference). Serious problems will attend any attempt to actually land equipment on the surface of Mercury, however. The temperature at the subsolar point, when Mercury is at perihelion, is about 440°C, which is more than hot enough to melt lead. The best study technique for the hot side will therefore include photography (preferably stereo photography), reflectivity and polarization measurements at various frequencies, and temperature mapping.

The dark side of Mercury, on the other hand, is probably the coldest place in the solar system. It has been billions of years since any solar radiation has fallen on this area, and Mercury does not possess an atmosphere to convect any heat around from the hot side. Indeed, the dark side must represent a steady-state condition achieved between heat radiated and heat conducted from the interior. The temperature of the anti-solar point is probably a few degrees absolute. An actual temperature mapping of the dark side would be very informative though difficult to achieve for such low temperatures. If Mercury possesses any remnant of its primordial atmosphere, that remnant should be present on the dark side as a solid. Again, analysis would be very desirable, but soft-landing equipment without both vaporizing the atmosphere and freezing the equipment into uselessness presents a real challenge.

Because of the eccentricity of its orbit (0.20562), Mercury experiences large librations in longitude. The resulting "twilight" zone is 47.4-deg wide. Mercury has a sidereal period of revolution of 87.9693 mean solar days; therefore, near the middle of the twilight zone, the sun slowly rises for about 3 weeks, then slowly sets for about 3 weeks at the same spot from which it rose; then there are about 6 weeks of darkness. It is within this twilight zone that landed experiments of the type used for surface study on the other planets are most likely to succeed.

Another useful satellite function is obviously that of magnetic-field mapping (if any field exists) and study of trapped cosmic rays (if any). Some useful data on these items as well as diameter measurements, photography, temperature measurements, and, possibly, a mass measurement could be obtained on a near miss, although a satellite is usually preferable where possible.

Sub-life forms are the most that can be hoped for in the case of Mercury. Biological considerations will be essentially the same as for the moon.

Special engineering preparations for Mercury flights will lie primarily in the area of temperature control in extreme radiation and conduction environments.

2. Jupiter. It would be difficult to imagine two bodies more radically different than Jupiter and Mercury. Jupiter has a diameter of 139,760 km, which is 10.97 times that of the earth. Its mass is 318.35 earth masses, and its mean density 1.33 g/cm^3. It travels about the sun in an orbit with a 5.202803-A.U. semi-major axis, requiring almost 12 years to make one revolution. (The sidereal period is 4332.588 mean solar days.) Jupiter has an extensive cloud covering which completely hides the surface. The rotational period of these clouds is from $9^h\ 50^m$ to $9^h\ 56^m$. They do not rotate as a rigid mass (see Fig. 60).

Methane and ammonia have been detected in the Jovian atmosphere in amounts equivalent to 150 meters and 7 meters of gas, respectively, at 0°C and 760-mm pressure. The measured mean molecular weight of the upper atmosphere is 3.3. Therefore, the greatest part of the atmosphere must be hydrogen and helium, elements whose spectra are unobservable from earth because of our atmosphere. There is other indirect but strong evidence that hydrogen and helium are the major constituents. The temperature at the top of the cloud layer is about 168°K (−105°C) according to a combination of observation and theory. Theory would indicate that the clouds themselves are composed primarily of ammonia crystals, or "ammonia cirrus" as Hess has called them. A total atmospheric depth between 100 and 120 km is indicated theoretically.

Knowledge of Jupiter's mass, moments of inertia, mean molecular weight, and temperature has allowed theo-

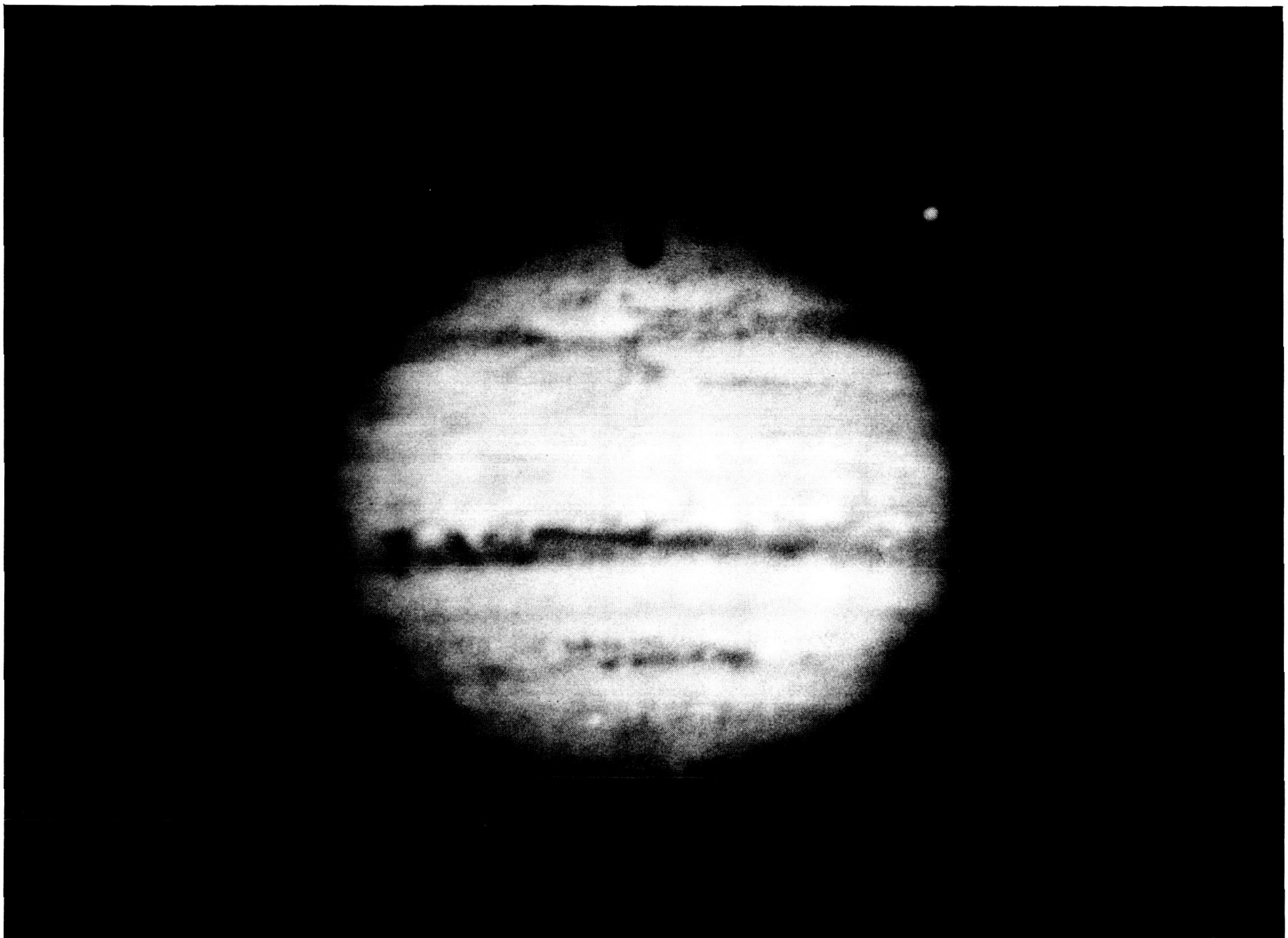


Fig. 60. Jupiter, Showing Ganymede and its Shadow (Photographed in Red Light Through the 200-in. Telescope)

Mt. Wilson and Palomar Observatories

retical models to be calculated for the Jovian interior. One very important factor in these calculations is the change in phase of hydrogen at a pressure of 800,000 atm. At this pressure, the density increases from 0.35 g/cm^3 (the molecular solid) to 0.77 g/cm^3 (the metallic solid). These figures quoted are strictly for absolute zero temperature but change only a negligible amount up to 10,000°K. The best representation of Jupiter, by Ramsey and Miles, consists of a core containing 5% of the mass consisting of helium, and small amounts of the heavier elements surrounding the other 95% of the mass consisting of hydrogen and helium in the ratio 22 to 1 by number. In this model, the metallic phase sets in at 60,500 km from the center. This model cannot be taken as exact reality, of course, but it does show the general type of structure which must be involved.

Other items of interest on Jupiter include the semipermanent markings and a radio source. The famous "red spot," most famous of the semipermanent markings, has been observed closely for 80 years, and descriptions in old observing records indicate a probable age of at least 300 years. The best suggestion to date is that small amounts of sodium in solution in liquid ammonia can produce the colors involved at the temperature of the Jovian clouds. Whether or not this is the correct explanation and why the marking seems so permanent is not known. A number of other markings of similar but lesser prominence have been noted through the years.

The radio source mentioned was discovered by Burke and Franklin in 1955 with equipment sensitive to 22.2 mc. The same source was soon discovered on earlier Australian records made at 18.3 mc. Further work soon indicated that three white South Temperate areas were probably responsible for the emission with the possibility that the red spot is an independent emitter.

A number of experiments are possible candidates for a Jovian probe. For a near miss or a Jovian satellite, the following experiments might be considered:

a. Magnetic-field and cosmic-ray trapping
b. High-resolution spectrograms of distinct regions of the Jovian atmosphere (particularly the red spot)
c. Photos (various colors and polarization) penetrating to different depths of the atmosphere (special attention to red spot)
d. Nature of Jovian radio emission (intensity vs location, and wavelength)
e. Temperature mapping
f. Radar-depth of atmosphere
g. Nuclear magnetic resonance using planet's field
h. Rotation of plane of polarization of radar beam
i. Electric fields
j. Atmospheric turbulence, etc.(Jovian meteorology)
k. Higher order terms in gravitational field

A semicontrolled penetration of the Jovian atmosphere might study some of the following problems:

a. Atmospheric pressure and density
b. Surface photographs
c. Mass spectrograph
d. Atmospheric extinction of light from sun and/or stars
e. Radio propagation through atmosphere
f. Temperature
g. Local electrical and magnetic phenomena

Problems involved in an actual landing on the surface of Jupiter are far enough in the future that they can be ignored at this time. The intense Jovian gravitational field (the surface gravity is about 2.64 of earth) will demand immense retro-rockets to accomplish even the semicontrolled penetration.

Another item of extreme importance is the Jovian satellite system. Jupiter, with 12 known satellites, has a miniature solar system of its own. Adequate study of these satellites will require an unfortunately large number of accurately guided probes. The items of greatest interest include masses, diameters, periods of rotation, nature of surface (regular or irregular), magnetic fields (especially of the 4 larger satellites), and residual atmospheres of the larger satellites. Probably a number of Jupiter's small moons are captured asteroids. It is, in fact, very difficult to understand how the retrograde satellites could be anything else. Satellites formed from the original Jovian protoplanet might be expected to have a regular shape and regular surface features. Captured asteroids would probably be very irregular. If the capture took place fairly recently, the new satellite might be rotating; whereas, primordial satellites may well have the same period of rotation and revolution. All of these items have a strong bearing on theories of the origin and evolution of the solar system.

3. ***Saturn.*** Saturn differs from Jupiter chiefly in possessing a very amazing and beautiful system of rings (see Fig. 61). Atmosphere and interior differ from Jupiter only in quantitative detail. All Jovian experiments are therefore Saturnian experiments, except study of a known radio source and the red spot (Saturn had a "white spot" which was seen from time to time up until a few years ago).

Experiments involving the rings could include:

a. High-resolution pictures (resolving individual particles) in order to obtain some knowledge of particle size and distribution
b. Mass of the ring (a completely unknown quantity) by interaction of ring and probe
c. Move through the rings (they are only 10 miles thick)
d. Move with the rings (when guidance and control permit)

Both c. and d. should include particle counts, or smashes, and analysis of captured debris by *X*-ray spectroscopy, *X*-ray fluorescence, etc.

The Saturnian satellite system of 9 members differs from the Jovian system mainly in that it is more regular and much more nearly a copy of the solar system. Saturn has only one retrograde satellite, the outermost member, Phoebe. Since asteroids do go out as far as Saturn, Phoebe may be captured. Actual experiments to be performed

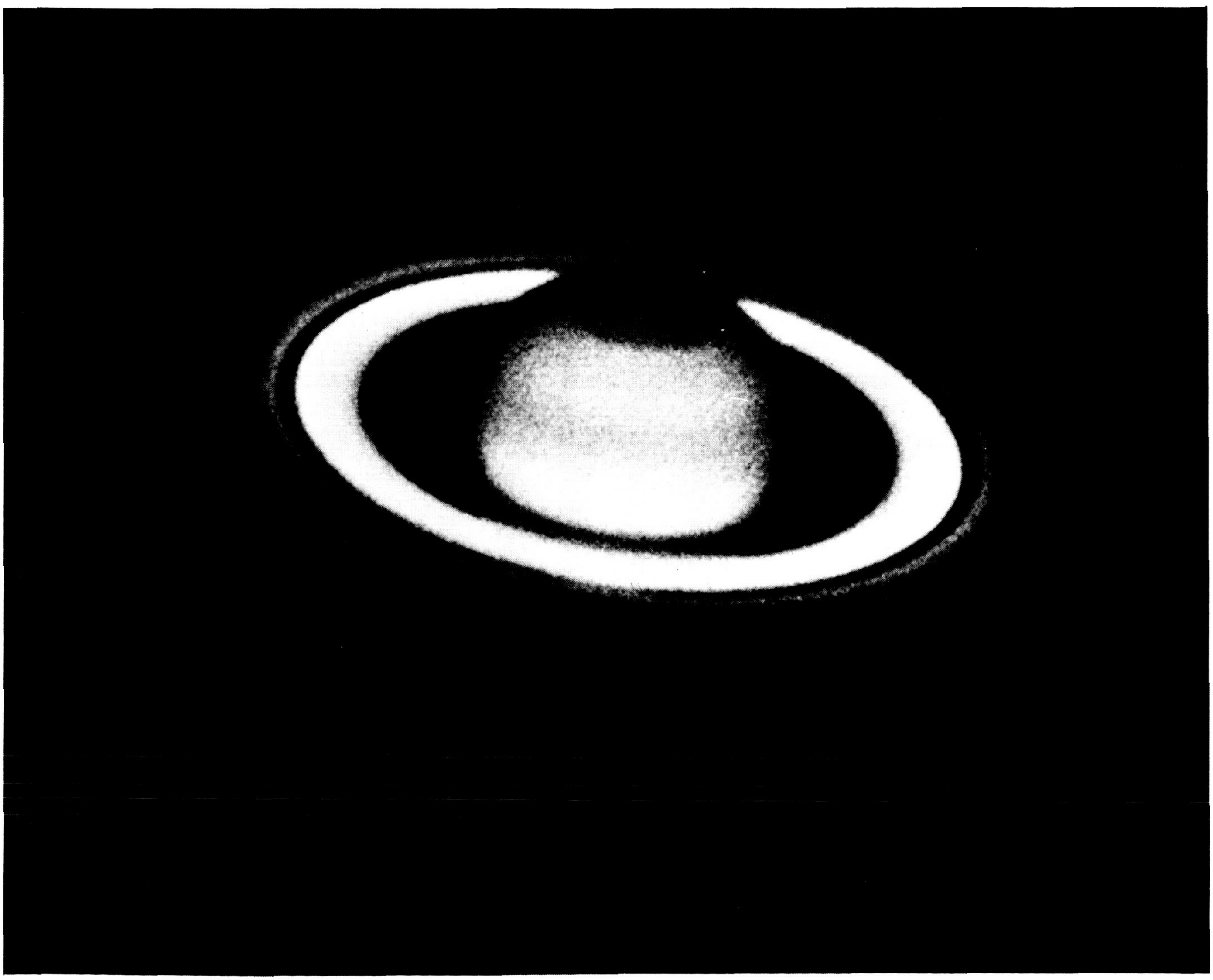


Fig. 61. Saturn
Lick Observatory

are the same as those in the Jovian system except that one satellite, Titan, is known to have an atmosphere, which should be studied.

4. Uranus and Neptune. Uranus and Neptune are merely smaller (about half the size) and more distant versions of Jupiter and Saturn, possessing 5 and 2 known satellites, respectively. At such a time that vehicle advances allow flight paths requiring considerably less time than minimum energy paths (16 and 31 years for Uranus and Neptune), experiments similar to the Jovian experiments may be flown to these planets, too.

5. Pluto. Pluto is an enigma. Almost nothing is known about it except its orbit and its period of rotation. The mass (determined from perturbations of orbits of Uranus and Neptune) and the apparent diameter imply a mean density of more than 50 g/cm^3, which is not possible. Either mass or diameter or both must be in serious error. A new measurement of angular diameter is definitely indicated, and this measurement should be made outside the earth's atmosphere, for Pluto's apparent angular diameter is only about ½ sec of arc. Perhaps an interferometer will offer a more accurate solution. New dynamical mass calculations must be attempted as Pluto's orbit becomes better known. It is unfortunate that a direct probe investigation of Pluto will be impossible for decades.

F. Asteroids, Comets, and Meteor Streams

1. Asteroids. The asteroids (minor planets or planetoids, as they are sometimes called) are fragments of matter which revolve about the sun mainly between the orbits of Mars and Jupiter. There is a possibility that many undiscovered bodies may be beyond Jupiter, although only a few are now known. There are also a number of asteroids with perihelion distances less than 1 A.U. One asteroid, Icarus, approaches within 0.19 A.U. of the sun. Although some similarities in orbital characteristics exist between asteroids and comets, the similarities end there; asteroids should not be considered as relatives of comets.

As discoveries of asteroids mounted into the hundreds toward the end of the 19th century, astronomers thought that perhaps a planet between Mars and Jupiter had been pulled apart by Jupiter or had run into another body, such as a Jovian satellite. More recent theories of the origin of the solar system, the von Weizsäcker-Kuiper theories and their modifications, doubt that one large planet ever existed between Mars and Jupiter. Kuiper feels that possibly as many as ten little planets were originally formed in this region and that most of the asteroids are the result of a collision between two of these original bodies and subsequent collisions among the fragments.

Interest in the asteroids lies in three major areas. First, any theory of the origin of the solar system must have a reasonable explanation for the origin and evolution of the asteroid belt. Second, the belt presents a tremendous challenge to celestial mechanics. Third, the belt is the probable source of the meteors, dust, and general debris cluttering up the solar system.

Probes could advance asteroid studies in several ways (none of them particularly feasible economically, however). There are various questions that could be investigated regarding the shape and nature of the largest asteroid, Ceres: Is it spherical or is it a fragment? Is the surface like the surface of the moon? Is it dust covered? What are the reflective and polarizing properties of the surface? Similar questions are in order concerning the other large asteroids. The answers have obvious bearing on the origin and evolution problem.

Hirayama has presented definite evidence of asteroid families, presumably resulting from fairly recent collisions. It would be very important if even one of these collisions could be dated by nondynamical means. Probably relative ages could be gained through measurement of the thickness of dust layers on asteroids passing through similar regions of space. A more quantitative measurement might be obtained through the measurement of surface radioactivity.

It has been estimated that there are 55,000 asteroids brighter than the photographic magnitude 19.5. No real estimate can be made of the number of still fainter (smaller or more distant) objects. Such an estimate is badly needed if we are to obtain a reasonably accurate value for the mass of the asteroid belt. If a mass spectrum could be obtained for the smaller particles, some estimate of intermediate objects could be made by interpolation, and a better value of the belt's mass obtained.

Research of this type will require a large number of probes and will be much more practical when extraterrestrial bases have been established to serve as launching points. A few opportunities do exist which are not quite so far in the future. One such opportunity involves the previously mentioned asteroid Icarus.

The orbit of Icarus is the most eccentric of any known asteroid; the eccentricity $e = 0.83$. The semimajor axis a has a value of 1.08 A.U. Thus, during a period of 1.1 years Icarus approaches within 18 million miles of the sun (half the distance of Mercury), recedes beyond Mars, and returns. The orbit of Icarus is inclined 23 deg to the ecliptic so that it actually never approaches the earth more closely than about 4 million miles (6.4×10^6 km), that particular configuration occurring about once every 18 years. Icarus can also approach within about 8 million miles (12.9×10^6 km) of Mercury. The next very favorable approach to the earth will occur in June of 1968. Serious thought should be given to taking advantage of this approach.

A landing on Icarus could serve two purposes. First, it could afford the first opportunity for direct study of an asteroid. The near-miss type of experiments previously discussed could be supplemented by gamma-ray spectroscopy, X-ray fluorescence, and, in fact, the whole battery of techniques previously perfected for lunar study. The place of asteroids in the scheme of the solar system could become far better understood. Second, Icarus could serve as a space vehicle for solar research. True, no propulsion energy would be saved by such a procedure, but the probe would then be at a known position in space (the orbit of Icarus is well known) and the asteroid might offer the probe some protection during the blazing perihelion passage. Even with protection, the probe might not survive such a passage, for the surface temperature of Icarus exceeds 500°C at that time.

Landing on Icarus would present problems. Icarus is only about 1 mile in diameter. From the propulsion standpoint, soft landing is simplified to a degree by the small gravitational attraction, escape velocity being only a few feet per second; but guidance would receive an extremely severe test. Taking all facts into consideration, an attempt definitely seems worthwhile.

A second object which may some day be of interest is Hidalgo, the asteroid with the largest known orbit. Hidalgo's semimajor axis is 5.80 A.U. An eccentricity of 0.656 means that Hidalgo comes near to the orbit of Mars and recedes almost to the distance of Saturn. The orbit of Hidalgo also has the greatest inclination of any known asteroidal orbit, 42.5 deg. Of course, the period of Hidalgo is also quite long, 13.9 years. This means that intermittent transmitter operation for a period of 7 years would be desirable. Also, with a perihelion distance of 2 A.U., guidance will be difficult until probes can be launched from Mars.

A third object of possible interest is Eros (No. 433) which can come within 14 million miles (22.5×10^6 km) of the earth at a favorable perihelion passage. The mean distance of Eros from the sun is 1.458 A.U., and the eccentricity of its orbit is 0.223. Its orbital inclination is 10°49′. The sidereal period of Eros is 643.2 days; its synodic period is 845 days. Thus we have another body which comes relatively near to us and could serve as a vehicle to the asteroid belt, for at aphelion Eros lies well within the belt. The next very favorable approach occurs in 1975. Relatively good approaches will occur sooner.

The roster of interesting objects is not exhausted with the three asteroids which have been discussed, but asteroid vehicle requirements are fairly well delineated by them. Studies to be made from these vehicles, other than on the asteroids themselves, depend upon the regions of the solar system traversed by them. Each proposed use of an asteroid as a vehicle will have to receive individual discussion. In many cases the advantages of an accurately known position will not outweigh the problems of an asteroid landing, and the probe will be sent out "on its own."

2. Comets. For hundreds of years comets were objects of superstitious awe. Then in 1577, Tycho Brahe noted that the great comet of that year exhibited no sensible parallax when seen from different parts of Europe and therefore must be something more distant than a cloud in the upper atmosphere. In 1705, Edmund Halley published his treatise on comets in which he applied the new Newtonian theory of gravitation. Nothing fundamentally new in the theory of comets appeared from that time until almost the turn of the present century when astronomers began applying spectroscopy to comets and physicists discovered that light could exert pressure, thus apparently explaining the fact that comet tails almost always trail away from the sun. In 1951, Whipple published his "icy conglomerate" theory of comets, which can qualitatively explain all observed cometary phenomena. Comparatively little quantitative information is available on comets, even today.

In the icy conglomerate theory, the nucleus of a comet is depicted as a very porous mass of solidified gases with a comparatively small amount of meteoric debris included in the mass. On the surface there may perhaps be a layer of dust picked up during slow passage through outer parts of the solar system. As a comet approaches the sun and is heated, the gases vaporize and stream back, forming the coma and tail (see Fig. 62). Comets survive for many perihelion passages because of the very poor heat conduction in the spongy nucleus. The irregular nature of the nucleus accounts for the sudden jets sometimes seen (as a pocket of gas is released) and for the irregular accelerations which sometimes make a comet several hours early or late in returning to a given point in its orbit.

The most important gases included in a comet are CH_4, NH_3, H_2O, and possibly CO_2 and C_2N_2. As a comet approaches the sun, these gases vaporize one by one and are broken up into radicals and ions by absorption of sunlight. Observed species include C_2, CN, CH, NH, OH, CH_2, HN_2, CH, N_2, and OH. Comets that approach very near the sun also begin to show lines due to sodium, iron, nickel, and other metals (which are present in the solid particles).

These are some of the facts we know. What we lack is quantitative information. First, no one knows the mass of a single comet. Only the upper limit is known, which would certainly be less than 10^{-4} the mass of the earth, because no comet has ever perceptibly affected the motion of any moon or planet, even when passing only a few tens of thousands of kilometers from them. Second, the

Fig. 62. The Head of Halley's Comet (60-in. Telescope, May 8, 1910)

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size of the nucleus is small, but no one knows how small. Comets passing directly in front of the sun completely disappear from view, implying that their bodies are not large. Probably the nucleus is not one mass but a combination of several, or even many separate masses. That no single mass is larger than a few kilometers is all that can be said. Third, there is no accurate knowledge about the relative amounts of dust and gas or even about the relative amounts of various gases. Particle and gas densities and ionization throughout the coma and the tail would be valuable knowledge.

One of the most serious cometary problems today is that of explaining quantitatively why comet tails always point away from the sun. For years radiation pressure was thought to be an adequate explanation. It is adequate for some types of tails (those known as Type II and Type III). Observed accelerations in Type I comet tails require repulsive forces of from 200 to 2000 times the attractive force of solar gravitation. Radiation pressure is totally inadequate to account for this. There is strong evidence that solar corpuscular radiation is responsible, but lacking quantitative data, the interaction mechanism must remain rather speculative.

Questions concerning the origin of comets, although a major field of interest, may be difficult to answer by probe study until travel far beyond Pluto is possible. The present state of comets, as well as their evolution, is a field that can profit greatly by probe study, however. First, because of the very low density of comet tails and comae, a probe could fly up the tail of a comet, unharmed, and collect density and ionization data. Second, a probe should be able to measure the nature and strength of solar corpuscular radiation interacting with a comet. Third, a probe should be able to measure the mass of the nucleus directly by orbiting it, although guidance into such an orbit would be very difficult. Fourth, a picture of the nucleus could be transmitted. Fifth, a small mass spectrograph should some day be capable, at least partially, of quantitatively analyzing the tail gases.

Finally, an actual landing on a cometary nucleus could be attempted. By using low-temperature gas jets as retro-rockets, disruption of the nucleus would be prevented. The use of one pure gas, such as pure helium, would prevent the landing area from being contaminated beyond hope of analysis. The landing vehicle would have a severe conduction problem to contend with, but proper construction should allow it to work for a reasonable time. The primary job on the surface would be chemical analysis and detailed photography.

It must be noted that it is not necessary to wait for a comet to "just happen along." There are many periodic comets with well-known orbits which pass reasonably near the earth. Encke's comet has a period of only 3.30 years, although it is very faint and perhaps not too satisfactory. A much better object of study might be the Giacobini-Zinner comet with a period of 6.59 years and a perihelion distance of 0.996 A.U. This comet was responsible for the tremendous meteor shower of October 1946 and will probably present another in late 1959. In recent years the earth has been in approximately the right position in its orbit for this phenomenon to occur at every second perihelion passage of the comet.

Although comet nuclei are small, the coma of an average comet is several tens of thousands of kilometers in diameter, whereas the tails are many millions of kilometers long (the record is about 300 million kilometers). Therefore, guidance into the tail or even into the coma should be simple, whereas guidance to the all-important nucleus may prove very difficult. In spite of possible difficulties, some comet study is certainly deserved after the most important planetary studies are well under way.

3. Meteor streams. Nearly everyone has seen a bright meteor (popularly called a shooting star) flash across the sky. Today most people know that these meteors are just small pieces of cosmic debris, bits of rock, which burn up because of friction when they enter the earth's atmosphere at high velocity. Many people have seen meteorites, the remnants of meteors so large that they were not entirely consumed during their fall. Many of these meteors are simply sporadic meteors, that is, meteors not particularly related in orbital characteristics or other recognizable phenomena to any other meteor. Other meteors appear in showers (perhaps one a minute, sometimes one or more a second) all radiating from one point in the sky for a period of a few hours up to several days. Orbits calculated for shower meteors show them to be bodies which have all been in one, comet-like orbit. A large number of meteors extended along a common orbit about the sun are said to be a meteor stream.

In 1862 and 1866, comets were discovered which had orbits essentially identical to meteors of the Perseid and Leonid showers. Astronomers began looking for other such coincidences and found them. It is now believed that all meteor streams are made up of cometary debris which gradually spreads out in orbit behind the comet. Long after a comet ceases to exist (because of exhaustion of gases) its meteoric material continues in the same orbit. Gradually, however, because of planetary perturbations, collisions, etc., the meteor stream is broken up. Some of the material contributes to the supply of sporadic meteors; more of the material is ground finer and finer by collisions and contributes to the general substratum of material which causes the zodiacal light and which is gradually swept into the sun by the Poynting-Robertson effect. Actually, of course, much of the cometary material is very fine to begin with and needs no reduction in size before it is swept into the sun.

Possibly the only probe experiment on meteor streams which might be undertaken is a study of the mass and size distributions of particles within the stream, together with an attempt to date the stream by some separate nondynamical means as suggested for asteroids. A charting of meteor streams in the solar system is certainly desirable from several points of view. (We never see a stream unless it intersects the earth's orbit.) A deliberate exploratory undertaking is hardly practical at the present time, however. Space is just too big.

G. The Interplanetary Medium

1. Charged particles.

a. Cosmic radiation. The term *cosmic radiation* properly applies only to particles which come to the earth from outside the solar system. The presence of this radiation is detected on the surface of the earth by the observation of secondary particles which are produced by chains of nuclear interactions initiated by the primaries at the top of the atmosphere. Attempts have been made to observe the primary radiation itself by getting above the atmosphere with balloons, rockets, and earth satellites. However, since balloons cannot go completely above the atmosphere, the experimental results obtained at balloon altitudes must be corrected for the flux of secondary particles as well as for the flux of albedo particles (both upward-moving secondaries and primaries which have been reflected by the earth's magnetic field). Data obtained at rocket altitudes are free of downward-moving air-produced secondaries, but must still be corrected for albedo particles as well as for secondaries produced in the rocket itself. At satellite altitudes the earth's radiation bands also become a problem.

Despite these experimental difficulties, quite a lot of information has been learned about primary cosmic radiation. The most recent information will be briefly reviewed here.

(1) Composition of primary radiation. The total number of particles arriving at the top of the earth's atmosphere over the magnetic poles is about 0.2 or 0.3 $cm^{-2} sec^{-1} sterad^{-1}$ at sunspot minimum, when the sun's influence is presumably the least. The radiation is primarily high-energy bare nuclei. Fewer than 1% of the primary particles are electrons, gamma rays, or neutrons. Most of the particles are protons, about 10% are alpha particles, and about 1% are heavier nuclei. The chemical abundance of these heavier nuclei in cosmic rays corresponds roughly, with a few important exceptions, to the natural abundance of elements in the universe.

The ratio of the amount of lithium, beryllium, and boron to the amount of carbon, nitrogen, and oxygen for cosmic rays is higher than the ratio for the universe as a whole by a factor of about a million. This is probably the result of the disintegration of heavier primary nuclei on collision with the interplanetary hydrogen. If such a process is assumed, and the amount of lithium, beryllium, and boron at the source is assumed to be negligible, it is estimated that the cosmic rays have passed through 1 to 5 g/cm^2 of matter in traveling to the earth from their source.

The heavy nuclei ($Z \geq 10$) are several times more abundant relative to the medium nuclei (carbon, nitrogen, and oxygen) in the cosmic radiation than in the universe as a whole. Singer suggests that this might be explained if the composition of the cosmic radiation at its source is mainly heavy ions ejected preferentially from certain types of stars, rather than ions distributed according to the universal abundance.

The ratio of carbon nuclei to oxygen nuclei is about ten times higher in cosmic rays than in the universe as a whole. This fact may be important in determining the nature of the stars which might be the original source of the nuclei.

(2) Energy distribution of primary radiation. Primary cosmic rays have been observed (either directly or indirectly) to have energies from 10^5 ev up to 10^{19} ev. In general the energy distribution goes as E^{-n}, where E is the total energy (kinetic energy plus m_0c^2) per nucleon, and n is between 1 and 2 and varies very slowly with energy. This holds for E greater than 10^{10} ev per nucleon.

At low energies there appears to be a cutoff for both protons and alpha particles. It is not known whether or not a similar cutoff exists for heavier particles. Furthermore, the high-energy end of the heavy particle spectrum is not well known, nor has the relative abundance of the elements in the cosmic radiation been determined as a function of energy.

(3) Time variations of primary radiation.

(a) Diurnal variations. The 24-hour variation is generally quite small. At sea level there is perhaps a 0.1% diurnal variation in the charged secondaries and a 1% variation in the neutron intensity. Since the ratio of neutrons to mesons produced is a decreasing function of the energy of the primary particle, the diurnal variation can probably be almost entirely attributed to the lowest-energy primaries. There is some evidence that the diurnal variation might be quite large for heavy ($Z \geq 10$) nuclei. This is still very uncertain; a properly instrumented satellite experiment should be able to resolve the problem.

(b) Variations of 27 days. Intensity variations corresponding to the 27-day rotation period of the sun are observed. At sea level the 27-day neutron variation may be 5 to 30%, while the charged-particle variation is about one-third as much. This indicates that the 27-day variation also affects mainly the low-energy end of the primary spectrum. There is evidence that at high altitudes and latitudes the effect may be 7 to 10 times larger than on the surface. The amplitude of the diurnal variation also shows a 27-day recurrence effect.

It is believed that plasma clouds are ejected from certain disturbed regions of the sun. This plasma has a very high electrical conductivity and thus probably has some of the sun's local magnetic field trapped in it. Because of the expansion of these clouds, hydromagnetic instabilities and turbulences form which in turn are capable of affecting the low-energy cosmic rays. There are several other theories to explain the diurnal and the 27-day variations, but they all depend on the interaction of the primary radiation with electromagnetic fields associated with the plasma flow from the sun.

(c) Forbush decreases. There are often worldwide cosmic-ray-intensity decreases which are generally, but not always, associated with worldwide magnetic storms. The intensity variations at sea level may be as large as 10%. At present there is disagreement about the energy dependence of this variation, but the plasma flow from the sun is again believed to be the culprit, even though these storms have no 27-day regularity.

(d) Solar cycle variation. The hour-to-hour and the day-to-day fluctuations in the cosmic-ray intensity are much greater when the sun is near maximum activity in its 11-year sunspot cycle. A much more surprising effect is that the average primary intensity is *inversely* related to the solar activity. During periods of sunspot maxima there is a great decrease in the number of low-energy protons reaching the earth; in fact, the total number of primary particles incident at the top of the atmosphere over the poles may differ by more than a factor of 4 between solar maximum and solar minimum. Despite the great fluctuations in the intensity of the cosmic radiation, it still remains essentially isotropic. Therefore, whatever the modulating medium is, it must be present out of the plane of the ecliptic as well as in it. One possible explanation is that during solar maxima when the plasma flux from the sun is large, the plasma clouds carry their trapped, turbulent magnetic fields to distances well beyond the earth's orbit. Low-energy cosmic rays have trouble diffusing through this great region of hydromagnetic turbulence to get into the inner solar system. A test of this theory would be to see if there really are no low-energy protons in the entire inner solar system at sunspot maximum. An alternative possibility, less widely accepted, is that the decrease is earth-centered and is due to interactions of the solar plasma with the geomagnetic field.

All the data given above are clues to the solution of the following two major puzzles connected with the cosmic radiation: (1) Where do the cosmic rays come from and how are they accelerated to such tremendous energies? (2) What is the mechanism of the interaction between the sun and the primary cosmic radiation? The importance of cosmic-ray theory in cosmology is demonstrated by the fact that the energy density in space due to the cosmic radiation is approximately equal to the energy density of starlight.

In order to solve the cosmic-ray puzzles, both the quality and the quantity of data on cosmic rays must be improved through the use of space probes. These probes should be instrumented to measure the composition and the energy distribution of both the primary cosmic radiation and the solar plasma flux and their variations with time, distance from the sun, and the local magnetic field (both magnitude and direction).

b. Solar flare particles. In the last 20 years there have been five instances of particles from solar flares being observed on earth. Of these, only the 1956 flare ejected particles with enough energy (more than 15 bev) to be observed at the geomagnetic equator. The energy spectrum is a lot steeper than the primary cosmic-ray spectrum as shown by the steeper latitude effect and the very high ratio (~ 50) of neutron increase to charged particle increase at sea level. In the early stages, the radiation arrives anisotropically from the sun. After reaching a maximum, this radiation fades away with a half-life of about 2½ hours. After the maximum of the anisotropic radiation has passed, an isotropic increase is observed which has a decay half-life of about 7½ hours. This behavior is considered to be further evidence for the existence of the region of magnetic turbulence beyond the earth's orbit already discussed in connection with the time variation of cosmic rays. Such a region must be capable of reflecting particles of more than 15-bev energy.

There are many questions still to be answered in connection with solar flare particles:

1. What are the charge and the mass distribution of these particles? Are there any electrons in these bursts?
2. Why are so many of the large solar flares not accompanied by the observation of particles on earth?
3. When these two questions are answered, then perhaps we can try to determine the mechanism by which particles are accelerated in flares.

c. Solar plasma flux. The importance of the plasma flux from the sun has already been pointed out in connection with the time variations of the cosmic radiation and the reflection of solar flare particles from regions beyond the earth. This same solar plasma flux probably also plays major roles in the generation of auroral and magnetic disturbances on earth and the formation of the earth's polar E layer and the curvature and ionization comet tails.

To date, all our knowledge of this important phenomenon has been obtained in an indirect manner. It is believed that this flux has both a steady and a burst component. The steady flux is thought to have a velocity of several hundred kilometers per second. Estimates of the particle flux range from 10^7 to 10^{12} cm^{-2}sec^{-1} at the earth's orbit.

The burst component is often, but not always, connected with visible solar disturbances. By timing the interval between the observation of the solar disturbance and the beginning of the associated geomagnetic disturbances, the particles have been assigned a velocity of between 350 and 2500 km/sec. However, if the particles did not come directly from the sun but had to perform essentially a "random walk" through the many magnetic-field inhomogeneities which may exist between the earth and the sun, then the actual particle velocity may be much higher than the values given above. The nature of strong aurorae also indicates a higher velocity particle. However, there is some controversy over whether or not auroral particles are accelerated after they reach the vicinity of the earth. The particle flux density has been variously quoted as being between 10^8 and 10^{11} cm^{-2}sec^{-1}.

Another problem which might be related to the solar plasma flux is the large discrepancy between the position and direction of the geomagnetic dipole as determined by magnetic-field measurements at the surface of the earth and as determined by the latitude variations of cosmic-ray secondaries. One theory is that the discrepancy is due to a distortion of the outer portions of the geomagnetic field by the solar plasma flow. Another theory is that the effect is due to the differences between the real geomagnetic field and the approximate dipole field.

It is clear that a space probe is necessary to measure the properties of this solar plasma flux. Its energy spectrum, its composition, its time variations, its direction of flow, and its associated magnetic fields must all be investigated.

d. Trapped plasma belts. The recently discovered radiation belts require much further investigation. The contours of these belts are roughly known. As one moves out from the earth at the magnetic equator, the radiation intensity begins to rise at about 900 km and keeps on rising to a peak at about 4000 km. Then the intensity decreases, followed by another rise to a second peak whose height and position vary with time. From here, the intensity decreases, but sometimes many small, narrow intensity peaks are superimposed.

The questions which must still be answered are: What are these particles—protons, electrons, or both? What is their exact distribution in space? What is their energy spectrum? How are they affected by solar activity? Once these questions have been answered, perhaps it will be possible to say where the particles in these belts come from. It may well be that the two major belts have very different sources.

2. ***Neutral interplanetary gas.*** It is not known whether or not there is any neutral interplanetary gas. The interstellar gas in the general neighborhood of the Milky Way has a density of about 1 hydrogen atom per cm^3. However, whatever gas is present within the solar system (away from the planets) might be ionized as a result of radiation from the sun.

3. ***Interplanetary dust.*** The density of dust in interstellar space corresponds to about 100 small grains (near 10^{-5} cm radius) in a cubic mile. In the solar system, especially in the plane of the ecliptic, the amount of dust is believed to be much greater. One reason for this assumption is the presence of the zodiacal light, which is a faint glow stretching along the ecliptic both ways from the sun. It has been traced completely around the sky photographically and photoelectrically.

Agreement is general that the zodiacal light is mostly sunlight scattered from dust. Since the zodiacal light shows a maximum polarization of about 20%, whereas light scattered from electrons is strongly polarized, electrons play only a minor role in it. Large numbers of scatterers of atomic or molecular size are ruled out by the lack of bluing of the scattered light.

During the lifetime of the solar system, all particles smaller than about 10 cm and within about three A.U. of the sun would have been swept into the sun by the Poynting-Robertson effect. Therefore, the material now present must be cometary or asteroidal debris which only recently attained its small size. The amount of dust present is thus important to studies of the evolution of the solar system in general and comets in particular.

Theoretical studies, as well as observations of meteors and the zodiacal light, lead to the following postulates about the interplanetary dust:

1. It is distributed according to the inverse 3/2 power of its distance from the sun.
2. The number of particles is proportional to the particle radius to the -3 or -4 power.
3. The minimum particle size is several microns.
4. Near the earth, there are probably 10^{-11} to 10^{-10} particles per cm^3 and a flux of 10^{-5} to 10^{-4} particles per cm^2/sec. *Pioneer I* found 10^{-6} particles per cm^2/sec, but it probably was not detecting the very smallest particles.

An interesting region of the sky which might be investigated with a Mars probe is the region of the gegenschein, or counterglow. The visible manifestation is a glow of very low surface brightness exactly opposite the sun in the sky. On a clear, moonless night, the glow is sometimes just visible to the naked eye. It has been photographed with fast wide-angle cameras and studied photoelectrically. It covers an area of perhaps 10 deg along the ecliptic by 7 deg wide. Its nature is unknown.

There are three theories of what the gegenschein might be. The oldest of these envisions the gegenschein as light reflected from small bodies trapped in the region of relative stability, which is one of the Lagrangian solutions to the earth-sun-particle three-body problem. If this is the case, there should be an increased density of dust material in a region 938,000 miles from the earth in the appropriate direction.

A second theory, proposed by Russian astronomers, concludes that the earth has a gaseous tail, just like a comet tail, which appears to us as the gegenschein. The third theory pictures the gegenschein as resulting from a special phase-angle effect of sunlight scattered from the anti-solar direction, with no increase in material above the general substratum of dust permeating the solar system.

All of these theories have their supporters and their detractors. A solution of the problem would not only answer the question of the origin of the gegenschein but would also present interesting evidence in other fields of astronomy (celestial mechanics, low-pressure gas dynamics, light scattering, etc.)

A probe equipped with a battery of micrometeorite detectors to measure the particle density as a function of distance from the earth, and with photocell-filter combinations to differentiate between reflected sunlight and the airglow-type of radiation the Russians say is given off by the tail, should be able to settle the whole matter.

4. ***Interplanetary electromagnetic fields.*** The determination of the properties of the interplanetary electromagnetic field is an important step toward finding solutions to many physical problems discussed in earlier Sections of this Report. Among these unsolved problems are:

1. The origin, acceleration, and propagation of cosmic radiation. Most theories in this field hinge on postulated interstellar and intergalactic electromagnetic field behavior.
2. The time variations of cosmic radiation as observed on earth. These variations are probably related to the electromagnetic fields associated with the solar corpuscular radiation.
3. The time variations of the geomagnetic field. These variations appear to be related to ionospheric currents and to the interaction of the solar plasma flux with the extremities of the geomagnetic field.
4. The discrepancy between the geomagnetic field as determined by surface field measurements and by the dependence of the cosmic-ray flux and spectrum on latitude and longitude. It is a matter of dispute whether or not this effect is due to the distortion of the geomagnetic field by solar and galactic magnetic fields.
5. Structure and evolution of the moon, the sun, and the planets. The sun's magnetic field has been estimated to be several gauss over much of the surface. However, there are fields of up to 5000 gauss near sunspots. Very little is known about planetary magnetic fields. An estimate has been made of the magnetic field of Venus, based on a statistical study of geomagnetic fluctuations during inferior conjunctions of Venus. This study led to the speculation that the magnetic field of Venus is strong enough to divert the solar corpuscular stream, and thus may be considerably stronger than the earth's magnetic field.

H. The Sun

The sun has been the object of more study than any other astronomical body, and for good reason. Besides being the nearest star, and thus the only star available for detailed study, it is the sun upon whose constancy of energy output all life in the solar system depends. Even so, much is not known about the sun. It is 98% hydrogen and helium. Yet the ultimate lines of these elements, Lyman α at 1216 Å and the strongest He resonance line at 591 Å, as well as the strongest He 2 resonance line at 304 Å, are completely hidden by the earth's atmospheric blanket which cuts off all optical radiation shorter than 2900 Å. The important Lyman discontinuity at 912 Å is hidden. In fact, the resonance lines of most astrophysically important atoms and ions lie in the hidden region. (Neutral alkali atoms, neutral and singly ionized calcium, and atoms and ions of some of the transition elements are the major exceptions.) Similarly, we have no knowledge of fundamental coronal lines, many of which lie in what is usually called the X-ray region. Observable coronal emission lines all result from high-lying metastable transitions.

The first information desired, then, is a tracing of intensity as a function of frequency, at the highest possible dispersion, from 2900 Å as far into the X-ray region as it is possible to go. Until dispersion sufficiently high to give detailed line shapes is possible, even simple line intensities are very worthwhile. As soon as it becomes possible to make a close approach to the sun, high-resolution scans of various parts of the solar disk for local variation will be both desirable and possible.

As a second type of experiment, spectroheliograms (pictures of the sun in the light on one spectral line) in the light of Lyman α, 519 Å of He, 304 Å of He 2, 2852 Å of Mg 1, etc., will be valuable. It goes without saying that the highest possible resolution is again desirable.

A third type of experiment will involve high-resolution direct photography whenever a close approach becomes feasible. At present, details of less than about one-half second of arc are generally unavailable [one-half second is about 360 km (224 miles) at the sun's distance]. (See Fig. 63.) Doppler shift and doppler broadening measurements of very small elements of the solar surface, especially around active regions such as sunspots, as well as in quiet areas, would be of great interest when close approach makes them possible. Nothing is known about small-scale motion on the sun.

Another broad class of experiments of importance will involve measurement of the general and local magnetic fields at the sun's surface and at the probe's location. The corpuscular radiation measurements discussed earlier should be made at the same time and correlation with field parameters attempted. If possible, a technique other than Zeeman splitting should be used for the surface field determinations.

The sun's corona must also be investigated. Twenty-five years ago, the solar corona was a complete mystery.

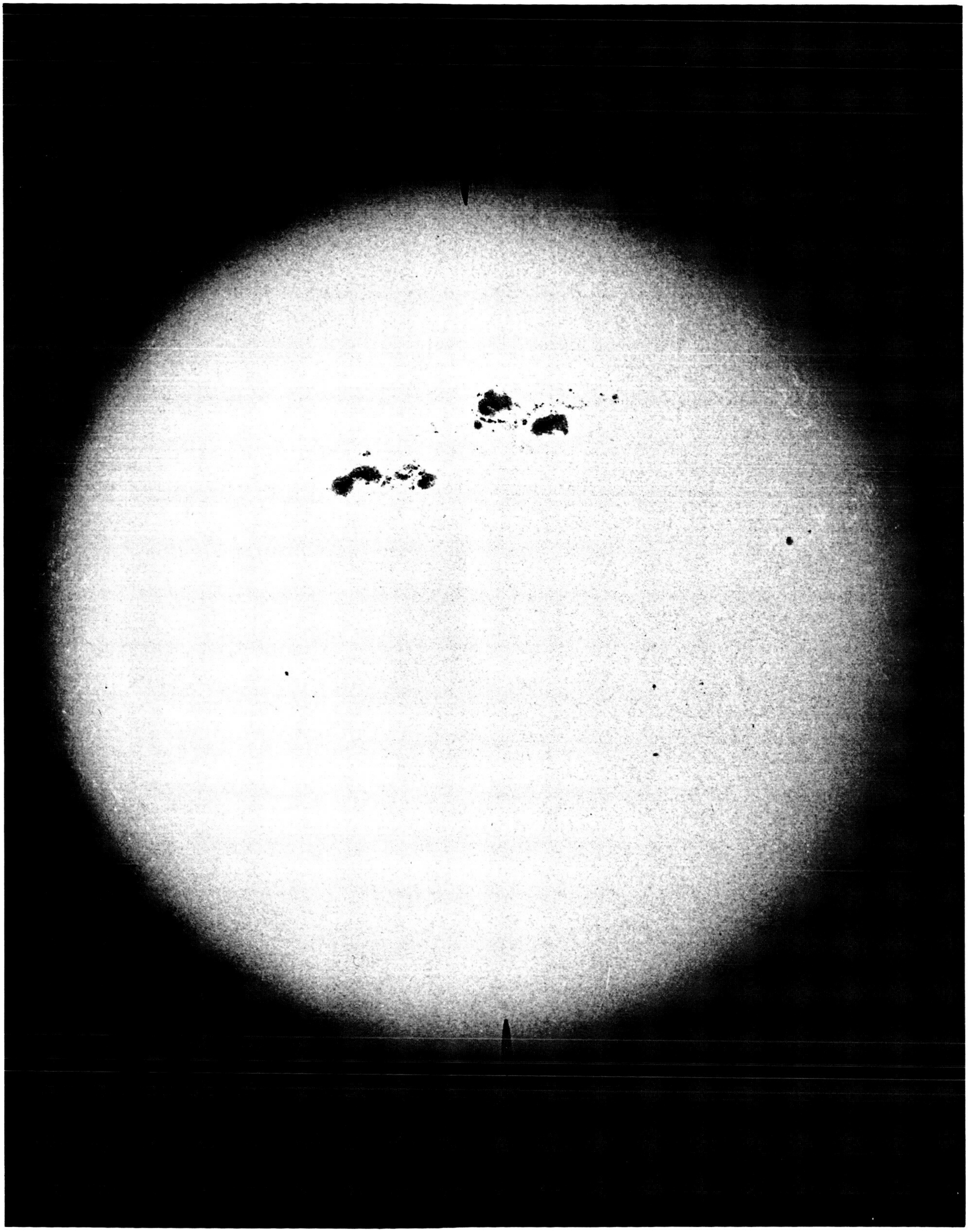

Fig. 63. The Sun, Showing Large Sunspots and Fine Structure of the Surface

Mt. Wilson and Palomar Observatories

Today we know what it is but not how it originates. In fact, the origins or causes of solar corona and the solar chromosphere form one of the least understood areas of solar physics.

The inner corona (out to about 1.3 solar radii from the center of the sun) has been described by van de Hulst as "a hot gas, virtually isothermal at a temperature of the order of a million degrees (Kelvin), in which most atoms are highly ionized. Its distribution shows certain structural details that keep changing continuously by small but definite streaming motions. The gas is made visible by the light scattered by the free electrons and by the spectral lines emitted by the ions, while superposed on these is the inner zodiacal light and the skylight. The light scattered by electrons (often called the K-corona) comprises 99% of the total light, has the same color as the sun, and is partially polarized."[8] The inner zodiacal light is solar light scattered from particles of relatively large size (dust and other debris), as is the case with the ordinary zodiacal light discussed earlier. These particles are not in equilibrium with the rest of the coronal matter but rather have low kinetic temperatures (velocities). The solar light reflected from these particles shows the regular solar Fraunhofer lines (from which this component of coronal light derives the name F-corona). The spectrum scattered toward the earth by the very rapid random motions of the electrons, on the other hand, shows only a continuum (except to the most careful measurement). The absorption lines have been broadened out into huge shallow features (more than 100 Angstrom wide) by the tremendous doppler shifts. There are 27 well-observed emission lines originating in the corona itself; the three brightest, which contribute about 1% of the inner coronal brightness, are caused by the ions Fe XIV, Fe X, and Ca XV.

The major difficulty with all observations of the corona is obviously the immediate proximity of the solar disk. The coronagraph, invented by B. Lyot in 1930, has made possible the study of the inner corona without an eclipse of the sun, but all observations are still hampered by the necessity of being made right through the bright daytime sky. The wave length dependence (λ^{-4}) of atmospheric scattering causes still more difficulty, for it makes the correction for skylight doubly difficult. The obvious solution is a coronagraph in space.

The coronagraph differs from a refracting telescope only in that it has an occulting disk to block out the sun's image, and special precautions are taken to minimize the scattering of light around the disk. (A simple lens is used, diaphragms are strategically placed, and clear air is sought.) Therefore, a coronagraph in space is no more difficult (nor any simpler) a problem than a telescope in space. At such time as the remote-control telescope in satellite or probe does become feasible, the remote-control coronagraph must also receive serious consideration; for the corona problem, the problem of how the corona is heated, of why it even exists, is felt by many to be one of the most important problems in solar physics. Theoretical understanding can come only when accurate quantitative statements replace the generalities of the preceding paragraphs.

The measurements mentioned are those broadly useful ones which will contribute to the solution of many solar problems (and, by extrapolation, to that of stellar problems). Many of these measurements can be made from earth satellites, but most, if not all of them, can be improved by actual approach to the sun. Second-generation experiments of critical importance will surely suggest themselves, following the preliminary surveys. Also, experiments of more limited interest, necessary for the solution of specific problems, have not been touched upon.

I. Supporting Research

During the past half century, only study of the stars has received a great deal of attention by the professional astronomer. There are a number of reasons for this, and one of the major reasons is the rise of "modern" physics. An understanding of the laws of interaction of radiation and matter, first empirically and then, with quantum theory, theoretically, naturally turned the interests of the astronomer to the previously enigmatic sun and stars. These same laws could have been applied to planetary atmospheres, but planets were the old and stars the new, and the new stellar astrophysics practically became astronomy. Simultaneously, the development of the special, and then the general, theory of relativity opened a new era in the field of cosmology, and interests went ever outward. The past twenty years have seen the application of nuclear physics, first to the theory of stellar

[8]van de Hulst, H. C., "The Chromosphere and the Corona," Chap. 5 of *The Solar System*, Vol I, *The Sun*, ed. by G. P. Kuiper. University of Chicago Press, Chicago, Illinois, 1953.

interiors and more generally to cosmology and stellar evolution. Astronomers, lost in the depths of interstellar and intergalactic space, have never returned to the planets.

There have been exceptions, of course. The observatory at Pic du Midi in France has contributed some excellent work during the past 25 years. Members of the Mt. Wilson staff have utilized the large reflectors of that observatory for planetary spectroscopy and radiometry, as have University of Chicago astronomers at McDonald Observatory. The Lowell Observatory has contributed what it could with mostly antiquated equipment and almost no funds. The new field of radio astronomy has made contributions. Very fine visual observations of the planets have been made by the amateurs of the world, especially in Great Britain and the United States. Yet the sad fact remains that no *modern* continuous research program on the planets has ever been undertaken.

There have been physical constraints on planetary research. Planetary research has suffered more than any other branch of astronomy from atmospheric opacity at critical wavelengths and from atmospheric turbulence, which destroys the fine definition of which astronomical instruments are intrinsically capable. Also, the realities of Newtonian mechanics tend to limit planetary study rather severely in the cases of Mercury, Venus, and Mars, three bodies of prime interest, while the back side of the moon is permanently hidden. The fact remains that much has been left undone.

It is self-evident that no probe study is scientifically justifiable if the same technical results can be obtained by terrestrial equipment, nor should any space exploration be undertaken without the fullest support of simultaneous earth-based research. Three problems of radically differing type illustrate these statements.

The moon offers a perfect example of research which has not been adequately done, research which in fact could not be accomplished by a probe in the foreseeable future, yet research needed for an adequate understanding of that body. The dynamical figure of the moon has been obtained from the Brown lunar theory and will soon be complemented by new values obtained with a lunar satellite. Equally important is the actual geometric shape of the moon. Until such time as a selenodetic net can be laid down on the moon in the manner of the geodetic surveys here on earth, these measurements of physical shape can best be made from earth. Yet such work has been only sporadically undertaken, the primary efforts being those of Franz (in 1889) and Saunder (in 1905).

Venus offers a problem whose solution should be attempted from earth before probe payload weight is wasted. Ultraviolet photographs of Venus show vague cloud markings. The observational problem is one of contrast. Modern television techniques are perfect for handling contrast problems, yet no such attempts have been made, even though a study of the clouds would offer a very good chance to solve the inclination problem and perhaps even the rotation problem. This information is badly needed for the probe program.

The value of simultaneous terrestrial support will perhaps be greatest for a Martian probe. Optical equipment is planned as an important part of the scientific payload of such vehicles. Yet recognition and interpretation of the famous "blue clearing" or even of a sandstorm could well prove difficult using "magnifications" totally different from those to which the astronomer is accustomed without simultaneous observations from earth showing the occurrence of these phenomena.

What about planetary research in a more general sense? How many of the facts quoted in our text books, facts upon which we are depending, rest upon one series of observations, perhaps made years ago with antiquated and inaccurate equipment or by an observer seeking to prove a pet theory rather than simply to discover the facts? Any new experiment in physics is soon repeated a dozen times by different people using different apparatus. An astronomical experiment done twice with order-of-magnitude agreement is considered well verified. The famous expression "astronomical accuracy" is a hangover from the days of dynamical astronomy and should be used in just the opposite of the usual sense when applied to most planetary astronomy or astrophysics. No astronomer denies this. It is the inevitable result of too few men, too little equipment, and too vast an area of study—the entire universe.

Another factor to be considered is the value of a continuous series of observations. Mars has usually been observed at opposition, Venus and Mercury near greatest elongation, and the other bodies of the solar system more or less randomly. The reasons for this are obvious. Observations of Mercury, Venus, and Mars at other than the times mentioned are difficult. Mercury and Venus are too near the sun, and Mars is too distant. Still, observations of Venus can be made at almost any time with proper equipment, and Mercury and Mars could be observed much more than they have been in the past. Such attempts should be made.

Time is the critical factor in a program of supporting astronomical research. Fortunately, only planetary spectroscopy really needs the light furnished by a large reflector. A good 36-in. Cassegrainian reflector (with Newtonian focus also available) will be adequate for many purposes. A wide-angle camera of moderate size would be useful for others. The success of the observatory will depend in large part upon the design of adequate auxiliary equipment and upon the size and enthusiasm of the computing staff.

This Report on the proposed space-probe program suggests such a supporting research program because of general agreement that supporting research, costing only a fraction of one per cent of the total space research funding, could well spell the difference between immediate success and a discouraging stretch-out in the over-all space program.

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VI. SUGGESTED PROGRAM

A. Flight Schedule

The flight schedule for the exploration of the planets depends upon both the availability of the equipment and the availability of the planetary target. The program discussed in this Report covers a time period from 1960 through 1964. During this time period, the principal vehicles to be used in the program are the *Vega* and the *Saturn*.

The targets considered to be feasible objectives for this time period are the moon, Mars, and Venus. The moon is available as a target every month. However, the planets are available only once for each synodic period. For Venus, this is approximately once every 19 months, and for Mars approximately once every 25 months.

In general, the suggested schedule calls for 3 flights a year during the 5-year period from 1960 through 1964, except for the year 1962, in which 4 flights are suggested, two each at Mars and Venus. At first glance, this seems like a meager firing schedule as compared with the usual missile-test schedules. However, it must be remembered that these flights do not include all of the flights which will be made with these vehicles. Several of the vehicles will be committed to satellite launchings during this same time period. In addition, it must be remembered that these flights are being carried out early in the development history of the vehicles, at a time when flight testing must proceed on a fairly cautious basis in order that maximum advantage may be taken of each flight test in further development of the missile. Furthermore, each of the flight tests will continue for several weeks or months, rather than for just a few minutes as for a typical military missile. The data resulting from this extended flight time must be given at least a preliminary analysis before sensible design decisions can be made for the next shot at that particular target.

Even as it stands, a schedule calling for 3 lunar or planetary shots every year for the next 5 years, using vehicles which have not yet had their first flight test, is extremely optimistic. It would be desirable to have backup firings available in the event the scheduled test met with disaster.

Unfortunately, the concept of a backup test takes on a new significance for planetary experiments. In order to meet the requirements for launching at a planetary target, the backup flight must be made within a few days of the original flight, or else delayed for a complete synodic period. Furthermore, since it will not be known for several months whether or not the payload satisfactorily completed its design objectives, it is not really possible to know within the few days allotted whether or not a backup is necessary. Thus, it would seem desirable to fire the backup shot regardless of whether or not the primary shot appeared to be a success within the first day or two.

B. Description of Typical Payloads

The payload weights available from the vehicles considered in this study are not yet accurately known. Very approximately, the *Vega* might carry 400 lb to Mars, 700 lb to Venus, and 1000 lb to the moon. Payloads for the *Advanced Vega* might be anywhere from 50% greater to twice as great. For the *Saturn*, preliminary estimates give values approximately 10 to 15 times larger than the *Vega*.

For many missions, large fractions of the total payload weight (all weight forward of the propulsion system of the last stage in the launching vehicle) must be devoted to retro-rockets or aerodynamic heating protection. Furthermore, the weight of the guidance system to be carried by the last stage must be subtracted from the gross payload before the true interplanetary payload is obtained. The weights of such systems are estimated to be about 100 to 200 lb, including power supplies necessary for their operation.

The weights of retro-rockets, mid-course correction rockets, aerodynamic heating protection, and structure can be estimated in terms of % payload weight. Such estimates are given in Table 5. As an example, assume the gross payload of the *Advanced Vega* for a Venus satellite mission is 1200 lb. First, subtract 150 lb for guidance of the launching rocket. Second, subtract 5% of the remaining 1050 lb for the mid-course correction rocket and at least 50% for the retro-rocket (assuming a circular orbit around Venus is required). This leaves 470 lb. The structure requires 15% of this, leaving 400 lb for other requirements. These other requirements include (1) attitude sensing and control system, (2) mid-course and terminal guidance systems, with target sensor, (3) telemetering instrumentation,

Table 5. Suggested Lunar and Planetary Flight Schedule[a]

Flights	1960			1961			1962			1963			1964		
Lunar Flights															
Near miss		V[b]													
Orbit						V[b]									
Rough landing					V[b]										
Soft landing											S[b]	S		S	
Orbit and return										S[b]			S		
Mars Flights															
Near miss			V[b]												
Orbit									S[b]						
Landing									V[b]						S
Venus Flights															
Near miss				V[b]											
Orbit								V[b]							
Landing								V[b]					S[b]		

[a]Legend: V, Vega; S, Saturn.
[b]Typical payload weight breakdown and list of potential experiments are given in Sec. VI-B.

Table 6. Weight Requirements of Payload Components

% of interplanetary payload[a]

Mission	Retro-Rocket, Including Structure[b] ($I_{sp} = 265$)			Aerodynamic Heating Protection	Structure % of net payload (subtract rockets and heating protection)
	Capture by Target, 2 Radii from Center	Circular Orbit, 2 Radii from Center	Landing		
Planetary miss	0	0	0	0	15
Lunar satellite	>15	>45	0	0	15
Venus satellite	>20	>50	0	0	15
Mars satellite	>50	>70	0	0	15
Lunar landing	0	0	>70	0	20-25
Planetary landing	0	0	0	35	20-25

[a]Mid-course correcting rocket accounts for 5% of total payload in all cases.
[b]Critically dependent on approach trajectory, i.e., time of flight (see Figs. 24, 25, 27, and 28).

(4) transponder circuitry, (5) communication power supply, (6) antenna, (7) miscellaneous power supplies, and (8) scientific instrumentation.

Since none of the components necessary for this particular payload has received an adequate design study, it is impossible to give meaningful estimates of their weights. On the basis of preliminary estimates, it is probable that between 50 and 100 lb might be available for scientific instrumentation for such a payload.

In this manner, estimates have been made of the total available weight for instrumentation in several of the payloads suggested in the proposed program. Tables 6 through 17 present listings of scientific instrumentation which might be included within such weight estimates for these payloads.

These Tables also show (1) the weight and volume requirements for each instrument, (2) the information rate or total amount of information which must be communicated to earth to give the results of the measurements, (3) the expected duration of the experiment, (4) the distance of the payload from earth at the time when the experiment might be performed, and (5) special remarks which apply to certain of the instruments. These listings are not the result of any official assignment of experiments to the various payloads. They represent only an estimate of reasonable experimental objectives consistent with present knowledge about the moon, the planets, and interplanetary space, and consistent also with present estimates of feasibility for both vehicle and payload design and instrument development. Thus, the information presented in these Tables should be considered as a starting point for more detailed program planning.

1. Lunar miss (Payload No. 1, August 1960). The first firing, in the summer of 1960, should be a "moon-miss" development test. This firing should test the scientific instrumentation, communications, guidance system, power supply, and attitude control system to be employed in the two subsequent firings. Table 7 summarizes the important characteristics of this probe.

2. Escape toward Mars (Payload No. 2, October 1960). This payload (see Table 8) will be fired with sufficient velocity to carry it to the orbit of Mars and will be launched at the right time and in the right direction to intercept Mars on its orbit. The guidance system available for the probe should have the capability of placing it within 1,000,000 miles of Mars.

The primary scientific experiment to be performed by this payload is infrared scanning of the region between 2 and 4 microns. This scanning will be accomplished by a spectrophotometer. In addition, a camera device (e.g., a vidicon) will photograph the same areas of Mars scanned.

If additional payload weight is available, a space package designed to measure environmental conditions between the earth and Mars will be included. Measurements of cosmic rays, magnetic field intensity, micrometeorite density, meteoric frequency, and solar radiation would be made by such a package.

The cosmic-ray instrumentation is designed to measure counting rate and total ionization by means of Geiger-Mueller counters and integrating ionization chambers. Several scaling circuits should be provided in case the probe should enter a high intensity radiation belt around Mars, if such exists. Some discrimination between protons and electrons in such a belt could be obtained if two similar ionization chambers were used, one with a stainless-steel wall and the other with a low-atomic-number material such as beryllium over stainless steel. If sufficient weight is available, information on the interplanetary primary cosmic-ray charge spectrums could be obtained using two pulse ionization chambers filled with gas at different pressure. This arrangement makes it possible to apply a correction for the background of nuclear stars produced in the gas. The charge spectrum and directionality of the relativistic radiation could be obtained by means of a counter telescope containing a Cerenkov counter, but it is doubtful that enough payload weight and communications capability will be available. The cosmic-ray detectors should preferably be on an arm out from the main body of the probe in order to minimize background due to showers originating in the material of the probe.

A magnetic-field measurement is useful both for itself and to help in the interpretation of the cosmic-ray data. A magnetometer of the alkali-metal-vapor type utilizing magnetic-resonance effects and able to measure fields between 1 and 10^{-4} gauss should be available. Such instruments available by this time will probably not be able to measure the direction of the field, but only its magnitude.

A measurement of micrometeorite density will provide valuable environmental data. Instrumentation to measure this would include microphones and erosion gauges. Instrumentation should be included for determining the

Table 7. Lunar Miss (Payload No. 1, August 1960)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Total Information bits	Experiment Sampling Rate	Duration of Sample	Duration of Experiment	Remarks
Optical system	vidicon, spectrophotometer	27	20 (peak) 6 (av)	7500	200,000 bits (total)	1/hr	3 hr	0 to 378×10^3	information can be stored for transmission after lunar passage when other information ceases to be of primary interest
Passive radar (far infrared)	scanner, infrared detector, pre-amp., osc., demodulator, tape recorder, temp. control, synchronizer	30	40	920	20,000 bits (total)	1/ 6 hr	1 day	0 to 378×10^3	
Gamma-ray spectrograph (U, Th, K)	scintillation counter, pulse-height analyzer (3 windows)	11	9	300	30	continuous	1 mo	378×10^3	
Cosmic ray	2 pulse chambers, 2 integrating chambers (one with beryllium over S.S. outer shell, other with S.S. outer shell), 2 Geiger-Mueller tubes, high voltage P.S.	12	2	180	2	continuous	6 mo	0 to 100×10^6	
Magnetic field intensity	alkali-metal vapor magnetic resonance magnetometer	5	5	60	2	1/day	6 mo	0 to 100×10^6	requires nonmagnetic environment
Micrometeorite density	microphones (and erosion gauges), circuits	1	2.5	25	—	continuous	6 mo	0 to 100×10^6	
Meteor detector	pressurized chamber with leak-rate detectors	3	0.1	120	2	continuous	6 mo	0 to 100×10^6	
Solar corpuscular radiation	scaling circuits, electrometer tube, condensor plates, Faraday cups, collimator	2	3	30	2	1/min or 1/hr	6 mo	0 to 100×10^6	instrument in open space pointing toward sun
Attitude sensor	photoelectric seeker	0.3	2	8	2	continuous	6 mo	0 to 100×10^6	
Timer		2	2.5 (peak) 1.2 (av)	60	—	continuous	6 mo	0 to 100×10^6	
Solar ionosphere	transmitter, antenna	—	5	—	2	1/ 3 hr	indefinite	—	

Total weight of instrumentation ....93.3 lb
Total weight plus structure ....112.0 lb
Total power (if all experiments are sampling at the same time) ...91.1 watts (peak)
Total volume ....9203.0 in.^{3}
Requirements: Injection guidance, attitude sensing and control, zero spin, minimum payload sterilization, 100×10^6 km telemetry.

Table 8. Escape Toward Mars (Payload No. 2, October 1960)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Information Rate bits/sec	Experiment Sampling Rate	Duration of Experiment	Distance from Earth km	Remarks
Optical system	vidicon, spectrophotometer	27	20 (peak) 6 (av)	7500	200,000 bits (total)	1/hr	3 hr	150×10^6	information can be stored for transmission after Mars passage when other information ceases to be of primary interest
Cosmic ray	2 pulse chambers, 2 integrating chambers (one with beryllium over S.S. outer shell, other with S.S. outer shell), 2 Geiger-Mueller tubes, high voltage P.S.	12	2	180	2	continuous	6 mo	0 to 200×10^6	
Magnetic field intensity	alkali-metal vapor magnetic resonance magnetometer	5	5	60	2	1/day	6 mo	0 to 200×10^6	requires nonmagnetic environment
Micrometeorite density	microphones (and erosion gauges), circuits	1	2.5	25	—	continuous	6 mo	0 to 200×10^6	
Meteor detector	pressurized chamber with leak-rate detector	3	0.1	120	2	continuous	6 mo	0 to 200×10^6	
Solar corpuscular radiation	scaling circuits, electrometer tube, condenser plates, Faraday cups, collimator	2	3	30	2	1/min or 1/hr	6 mo	0 to 200×10^6	instrument in open space pointing toward sun
Attitude sensor	photoelectric seeker	0.3	2	8	2	continuous	6 mo	0 to 200×10^6	
Timer		2	2.5 (peak) 1.2 (av)	60	—	continuous	6 mo	0 to 150×10^6	

Total weight of instrumentation ....52.3 lb
Total weight plus structure ....62.8 lb
Total power (if all experiments are sampling at the same time) ...37.1 watts (peak)
Total volume ....7983.0 in.^{3}
Requirements: Injection guidance, attitude sensing and control, zero spin, 200×10^6 km telemetry, maximum payload sterilization, local sensing of Mars.

danger due to meteors. Such instrumentation should be able to determine the probability of encountering a meteor large enough to penetrate a wall thickness comparable to that of a manned capsule. A gas-filled chamber of the proper wall thickness could be used for this purpose. A hit by a penetrating meteor could then be detected by monitoring the gas pressure in the chamber. A difficulty here is that in order to gather sufficient statistics such a chamber would probably have to be quite large. The cosmic-ray ionization chambers (which contain argon gas under pressure) might double as such detectors, but would be inefficient because of their small size.

3. Escape toward Venus (Payload No. 3, January 1961). This payload (see Table 9) will include all of the experiments described for Payload No. 2. The spectrophotometer, however, will be scanning a different region. The payload will also include a passive radar scanning in the far infrared region. This measurement will provide a rough measure of the surface temperature of Venus. Since the subsequent design of hard- and soft-landing probes for Venus is highly dependent on the planet's surface temperature, this measurement is given a high priority.

In addition, instrumentation will be carried for measuring the effects of the solar ionosphere on the transmission of radio signals. This experiment will continue indefinitely or until failure of the solar cells supplying the necessary power for transmission. This experiment will begin after Venus passage when the probe enters its final heliocentric orbit. The experiment will transmit a signal through the sun's ionosphere.

The characteristics of this probe are summarized in Table 9.

4. Lunar rough landing (Payload No. 4, June 1961). Sufficient gross payload weight should be available by this time to permit the use of a retro-rocket to slow the lunar rough-landing probe (see Table 10) to a velocity of about 100 to 200 ft/sec before impact and still leave ample payload available for instrumentation. It is not anticipated that a complete terminal-guidance system required for a soft landing (less than 50 ft/sec) will be available at this time. A cone penetrometer containing accelerometers can be used to measure surface hardness. This will require attitude control of the vehicle. The impact transmitter will transmit the impact data until it is destroyed by the impact. Such a surface-hardness measurement will provide valuable data for the lunar soft-landing probe. The cone penetrometer should be able to distinguish a surface of dust or sediment from one of hard rock.

If the landing is made on the dark side of the moon, the impact location could be marked by flash powder. The 200-in. telescope should be able to detect the flash of about 10 lb of powder if the telescope is aimed at the proper point. If the landing is made on the light side of the moon, close-up pictures of the lunar surface might be taken on the way in, giving an excellent view of surface features. A difficulty here is that large transmission bandwidth antenna would then be required to give sufficient data to transmit the pictures before the camera is destroyed on impact.

Experiments to be carried out after landing depend upon the development of instrumentation sufficiently ruggedized to survive the impact and the harsh temperature environment on the lunar surface. A ruggedized beacon and telemetry system with power supply, ruggedized temperature sensors, and a ruggedized seismograph can probably be developed. Also, it would be highly desirable to develop a radiation monitor and magnetometer to survive the impact. The lack of atmosphere and the probably small magnetic field on the moon should make it an excellent base from which to monitor cosmic radiation.

5. Lunar satellite (Payload No. 5, September 1961). This payload (see Table 11) will be placed in a well-controlled orbit around the moon using terminal guidance. Sufficient payload weight will be available for some rather elaborate instrumentation.

A gamma-ray spectrograph[9] will be able to compare the relative abundances of uranium, thorium, and potassium in the lunar crust. This measurement should make it possible to determine whether the surface of the moon is composed of granite, basalt or meteoric material. For a trajectory within 100 miles of the lunar surface, variations in surface composition could be determined with a resolution of approximately 100 miles.

High-resolution photographs of the surface of the moon will be taken at various wavelengths and polarizations. These photographs should provide information on the surface characteristics of the moon that will be valuable for choosing a site for a lunar soft landing.

[9]Proposed by James Arnold, University of California, La Jolla, California.

Table 9. Escape Toward Venus (Payload No. 3, January 1961)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Information Rate bits/sec	Experiment Sampling Rate	Duration of Experiment	Distance from Earth km	Remarks
Optical system	vidicon, spectrophotometer	27	20 (peak) 6 (av)	7500	200,000 bits (total)	1/hr	3 hr	75×10^6	information can be stored for transmission after Venus passage when other information ceases to be of primary interest
Passive radar (far infrared)	scanner, infrared detector, preamp., oscilloscope, demodulator, tape recorder, temperature control, synchronizer	30	40	920	20,000 bits (total)	1/ 6 hr	1 day	75×10^6	
Cosmic ray	2 pulse chambers, 2 integrating chambers (one with beryllium over S.S. outer shell, other with S.S. outer shell), 2 Geiger-Mueller tubes, high voltage P.S.	12	2	180	2	continuous	6 mo	0 to 100×10^6	
Magnetic field intensity	alkali-metal vapor magnetic resonance magnetometer	5	5	60	2	1/day	6 mo	0 to 100×10^6	requires nonmagnetic environment
Micrometeorite density	microphones, (and erosion gauges), circuits	1	2.5	25	—	continuous	6 mo	0 to 100×10^6	
Meteor detector	pressurized chamber with leak-rate detectors	3	0.1	120	2	continuous	6 mo	0 to 100×10^6	
Solar corpuscular radiation	scaling circuits, electrometer tube, condensor plates, Faraday cups, collimator	2	3	30	2	1/min or 1/hr	6 mo	0 to 100×10^6	instrument in open space pointing toward sun
Attitude sensor	photoelectric seeker	0.3	2	8	2	continuous	6 mo	0 to 100×10^6	
Timer		2	2.5 (peak) 1.2 (av)	60	—	continuous	6 mo	0 to 100×10^6	
Solar ionosphere	transmitter, antenna	—	5	—	2	1/ 3 hr	indefinite	—	

Total weight of instrumentation .... 82.3 lb
Total weight plus structure .... 98.8 lb
Total power (if all experiments are sampling at the same time) ... 82.1 watts (peak)
Total volume .... 8903.0 in.^{3}
Requirements: Injection guidance, attitude sensing and control, zero spin, 100×10^6 km telemetry, maximum payload sterilization, local sensing of Venus.

Table 10. Lunar Rough Landing (Payload No. 4, June 1961)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Information Rate bits/sec	Experiment Sampling Rate	Duration of Experiment	Distance from Earth km	Remarks
Surface hardness	cone penetrometer with impact transmitter	44.0	2	600	max. available BW	1/sec	at impact		
Communications	ruggedized beacon and telemetry	15	5	200	1000	continuous	2 yr	378×10^3	
Surface temperatures		1	0.1	12	100	continuous	2 yr	378×10^3	
Magnetic field intensity	ruggedized magnetometer	2	0.2	20	8	continuous	2 yr	378×10^3	
Radiation intensity	ruggedized radiation monitor	10	0.1	300	1	continuous	2 yr	378×10^3	
Impact location	flash powder to mark impact	10	0	150					
Seismic noise	seismograph	33.0	2	300	0.2	continuous	2 yr	378×10^3	
Photographs (detailed)	camera, scan circuits	10[a]	16 (peak) 6 (av)	200[a]	750,000 bits (total)				to replace flash powder

Total weight of instrumentation . . . . 135 lb
Total weight plus structure . . . . 162 lb
Total power (if all experiments are sampling at the same time). . . 25.4 watts (peak)
Total volume of instruments . . . . 1800 in.^{3}
Requirements: Injection guidance, attitude sensing and control, zero spin, maximum payload sterilization, retro-rocket, temperature control.

[a]Including container in which photograph is taken.

Table 11. Lunar Satellite (Payload No. 5, September 1961)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Information Rate bits/sec	Experiment Sampling Rate	Duration of Experiment	Distance from Earth km	Remarks
Gamma-ray spectrograph (U, Th, K)	scintillation counter, pulse-height analyzer (3 windows)	11.0	9	300	30	continuous	1 mo	378×10^3	
Photographs (high resolution, various wavelengths and polarizations)	camera, telescopic lenses, tape recorder, scan circuits, filters	55.0	120 (peak) 36 (av)	920	1080×10^3 bits (total)	1/ 15 sec	1 mo	378×10^3	designed to utilize the entire communications capacity
Magnetic field intensity	magnetic resonance magnetometer	5	5	60	3	continuous	1 mo	378×10^3	requires nonmagnetic environment
Mass spectrograph	logarithmic amplifier circuits, collecting device, mass spectrometer tube	22	17	700	90	continuous	1 mo	378×10^3	
Cosmic rays		20	4	300	—	continuous	1 mo	378×10^3	instrumentation dependent on whether or not the moon has a high-intensity radiation belt (to be determined by other shots)
Engineering development	attitude control, sensing, telemetry	150	50	1840	—	continuous	1 mo	378×10^3	
Attitude sensor		0.3	2	8	2	continuous	1 mo	378×10^3	
Timer		2.2	2.5 (peak) 1.2 (av)	60	—	continuous	1 mo	378×10^3	

Total weight of instrumentation ....265.5 lb
Total weight plus structure ....318.6 lb
Total power (if all experiments are sampling at the same time)..209.5 watts (peak)
Total volume of instruments ....4188.0 in.^{3}
Requirements: Injection guidance, attitude sensing and control, zero spin, terminal guidance, retro-rocket, maximum payload sterilization, transmitter interrogation.

A mapping of the lunar magnetic field and cosmic-ray measurements will be made to measure the radiation present in the vicinity of the moon. The cosmic-ray instrumentation chosen will depend largely on the information obtained from previous shots and upon whether or not the moon has been found to have a high-intensity radiation belt.

The mass spectrograph will attempt to determine the presence of a lunar atmosphere. The mass spectrometer will need a gas collection device in order to make a measurement, and even if such a collective device were developed, the lunar atmospheric density may be too low to be detectable in the background of gases produced by the outgassing of the probe itself. Such a mass spectrograph and associated collecting device should be tested in an earth satellite before this shot.

Observations of the orbit of the satellite will give a better determination of the mass of the moon.

6. Venus satellite (Payload No. 6, August 1962). The mission of this Venus satellite (see Table 12) and of the Venus entry shot which follows it are not only to acquire scientific data but to obtain environmental and engineering data that will be necessary for the design of the Venus soft-landing probe scheduled for 1964.

Ionospheric soundings of the Venusian atmosphere will be carried out with transmitters and receivers designed to operate at various frequencies between 500 kc and about 2000 mc. The main purpose of these soundings will be to find a radio "window" in the atmosphere and identify ionospheric layers. The receivers in this system will also detect sources of electromagnetic radiation (e.g., thunderstorms) generated on the planet.

Radar operating at the 3, 1.35, and 0.5 cm wavelengths will determine the relative abundances of oxygen and water vapor in the atmosphere. This equipment can also be used to give low-resolution radar mapping of the surface, with the resulting roughness measure a possible indication of the presence of life forms.[10]

Infrared devices will be employed to analyze the atmosphere and also to attempt infrared photography of the surface of the planet, particularly on the dark portion where the white vapor in the atmosphere may be absent. Photography will also be carried out in the visible and ultraviolet regions.

[10]Suggested by Gold, Space Science Board, National Academy of Sciences, Washington, D. C.

The cosmic-ray package will detect the extent of any trapped radiation in the vicinity of Venus and measure the primary radiation en route to Venus. A mass spectrograph will determine the constituents of the Venusian atmosphere at altitudes along the satellite orbit.

A magnetometer, probably of the magnetic-resonances type, will be included to give a mapping of the magnetic field of Venus.

7. Venus entry (Payload No. 7, August 1962). The Venus-entry package (see Table 13) will be equipped with sufficient protection against aerodynamic heating to provide a good chance of landing an instrument package on the surface. The entry vehicle will also include instruments to measure the characteristics of the Venusian atmosphere during descent.

A mass spectrograph will measure atmospheric composition. Thermocouples in the skin of the descending vehicle will give data making it possible to calculate the atmospheric temperature if the atmospheric composition and the Mach number are known. Light intensity near the surface should be measured by a photodiode to determine whether there is sufficient intensity to operate solar cells in the soft-landing probe.

At impact, a cone penetrometer (requiring attitude control) could determine the surface hardness. A ruggedized beacon with power supply designed to survive the impact and surface environment will be deposited on the surface. Tracking this beacon will give information on the rotation rate of the planet. If the beacon operates a sufficiently long time it can be used as a terminal-guidance device for the Venus soft-landing probe. This beacon will also contain sensors to give surface temperature.

Instrumentation to measure the moisture content of the atmosphere and to indicate whether or not the probe impacts on water should be included. This might consist of a hygroscopic resistor. An instrument for measuring the amount of dust in the atmosphere would also be valuable. A light source with mirrors to give a long light path, used in conjunction with a phototube to measure light attenuation, might be developed for this purpose.

Communications problems on this shot are particularly formidable and require considerable study. It is likely that a large vehicle antenna cannot be landed or even carried into the lower reaches of the atmosphere, which severely limits the available communications bandwidth.

Table 12. Venus Satellite (Payload No. 6, August 1962)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Information Rate bits/sec	Experiment Sampling Rate	Duration of Experiment	Distance from Earth km	Remarks
Ionospheric sounding and electromagnetic radiation	transmitter and receiver, antennas	40	200	1100	16	2/hr	2 mo	75×10^6	
Infrared photography	infrared vidicon, scanning device, preamplifier, and filters	9	20	500	30,000 bits (total)	1/ 3.5 hr	2 mo	75×10^6	8 shades of gray
Radar	high frequency (3, 1.35, 0.5 cm) transmitter and receiver, antennas	44	360	1200	16	2/hr	2 mo	75×10^6	
Photographs	camera, scan circuits, tape recorder, filters	26	20 (peak) 6 (av)	200	750,000 bits (total)	1/ 3.5 hr	2 mo	75×10^6	1.3×10^6 bits storage required 8 shades of gray
Cosmic ray	Cerenkov counter, ionization chamber, pulse-height analyzer, Geiger-Mueller tubes, high voltage P.S.	20	2	200	4	continuous	8 mo	75×10^6	
Mass spectrograph	collecting device (titanium band), mass spectrometer tube, logarithmic amplifier circuits	22	17	600	20	continuous	8 mo	75×10^6	
Magnetic field	magnetic resonance magnetometer	5	5	60	3	continuous	8 mo	75×10^6	requires nonmagnetic environment
Attitude sensor	photoelectric seeker	0.3	2	8	4	continuous	8 mo	75×10^6	
Timer		2.2	2.5 (peak) 1.2 (av)	60	—	continuous	8 mo	75×10^6	

Total weight of instrumentation ....168.5 lb
Total weight plus structure ....202.2 lb
Total power (if all experiments are sampling at the same time)..628.5 watts (peak)
Total volume ....3928.0 in.^{3}
Requirements: Injection guidance, attitude sensing and control, zero spin, terminal guidance, retro-rockets, 100×10^6 km telemetry, transmitter interrogation, maximum payload sterilization.

Table 13. Venus Entry (Payload No. 7, August 1962)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Information Rate bits/sec	Experiment Sampling Rate	Duration of Experiment	Distance from Earth km	Remarks
Atmosphere composition	collecting device, (titanium band) mass spectrometer tube, logarithmic amplifier, recorder	20	17	800	100	1/day	6 mo	0 to 75×10^6	continuous sampling when within 10,000 km of Venus
Temperature of atmosphere	thermocouple	0.5	0.1	16	10	1/sec	1 hr	0 to 75×10^6	
Light intensity near surface	photodiode	0.1	0.1	1					
Rotation rate of planet	rugged beacon, antenna, power supply	60	600	1500	—	2/hr	2 mo	0 to 75×10^6	1-min pulse duration
Surface temperature	resistance thermometer	1	0.5	20		—			
Surface hardness	cone penetrometer, low frequency accelerometer	44	2	600	max. available BW	—	1 sec (at impact)		
Timer		2.2	2.5 (peak) 1.2 (av)	60	—	continuous	6 mo	0 to 75×10^6	
Moisture content	hygroscopic resistor	1	0.1	20	5	1/sec	1 hr	75×10^6	
Dust detector	light source, mirrors, phototube	3	3	200	1	1/sec	1 hr	75×10^6	

Total weight of instrumentation . . . . 131.8 lb
Total weight plus structure . . . . 158.2 lb
Total power (if all experiments are sampling at the same time) . . 625.3 watts (peak)
Total volume of instruments . . . . 3217 in.^{3}
Requirements: Injection guidance, attitude sensing and control, zero spin, terminal guidance, 100×10^6 km telemetry, maximum sterilization.

Some interplanetary experiments could be performed en route to Venus, if weight limitations permit.

8. Mars satellite (Payload No. 8, November 1962). The payload and objectives of this shot (see Table 14) are very similar to those in the Venus satellite. The problem of finding a radio "window" in the atmosphere of Mars is probably much simpler than is the case with Venus. Ionosphere soundings can be made at lower frequencies. Also, more emphasis can be placed on obtaining photographs in the visible region because of the transparency of the atmosphere.

9. Mars entry (Payload No. 9, November 1962). The Mars-entry payload (see Table 15) will be similar to that of the Venus-entry payload. If sufficient communications bandwidth can be obtained, photographic equipment might be included to allow close-up photographs.

10. Lunar orbit and return (Payload No. 10, February 1963). The purpose of this test will be twofold: (1) It will demonstrate the capability of performing such a mission as flying around the moon and returning to make a safe landing on earth. (2) It will be a full systems test of the Venus-landing payload (payload 12).

The instrumentation will be similar to that scheduled for the Venus-landing package, except that it will be modified to the extent necessary for operation on the surface of the earth. If the Venus-entry shots or Venus satellite shots have determined that the surface of Venus is either largely water or land, this payload will be designed to operate accordingly, and will be brought into earth over either ocean or land as required. If this question is still unanswered, the selection of land or water operation will be based on the most reasonable estimate of Venus surface conditions from the available evidence. Experiments in the vicinity of the moon will be those which can be carried out with the equipment available as part of the system test.

11. Lunar soft landing (Payload No. 11, June 1963). This payload of the lunar soft-landing probe (see Table 16) is intended principally as a geological exploration of the moon's surface and would include a complete photographic survey of the landing area, the determination of the surface texture and composition (mineralogical and chemical) at various selected spots around the landing site, and a measurement of the more general environmental factors such as weathering (i.e., by radiation), seismic activity, and magnetic fields.

If possible, the surface exploration would utilize a mobile vehicle whose motion and experimental program would be controlled by commands from earth. First the camera would take a complete series of color photographs, ranging from long-range panoramas to microscopic examinations of the immediate surface. On the basis of these photographs, a suitable sampling site would be chosen and the vehicle ordered to proceed there. A vibrator probe would then determine the thickness and texture (i.e., dust or debris) of the surface layer and measure the thermal conductivity of the material by heating a thermistor at the probe tip and recording the subsequent cooling curve. Chemical analysis would be attained by X-ray fluorescence, ultraviolet emission spectra, and neutron activation, so as to yield data on both the light and heavy elements. A determination of the natural radioactive species will give some information as to age and geologic history. X-ray diffraction and infrared luminescence will provide a fairly complete mineralogical analysis. Surface density (by beta and gamma scattering), thermoluminescence, and volatile constituent analysis (by mass spectrometry of a pyrolized sample) will indicate the degree and type of weathering.

Some of these experiments can be performed merely by lowering the instrument onto the lunar surface, others will require sample collection and manipulation. The latter, although more accurate and unambiguous, are more subject to malfunction and unforeseeable hazards. Therefore a certain amount of experimental redundancy is desirable, especially in the chemical analysis.

The lunar environment presents some unique difficulties, chief of which are the high vacuum and temperature extremes. Not only will it be difficult to design moving mechanical systems which can function without lubrication in high vacuum, but it is highly probable that the dust will adhere to any surface with which it comes in contact, e.g., camera lenses. The wide range of temperature extremes makes it seem unlikely that all of the payload can perform satisfactorily during both the lunar day and night. Considerations of power dissipation and moving mechanical parts tend to favor the lunar night, whereas the presence of light for the all-important photographic survey and for solar cells (which could greatly reduce the weight of the power supply, and thereby make a large roving vehicle more feasible) makes the lunar day seem more favorable.

Table 14. Mars Satellite (Payload No. 8, November 1962)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.³	Information Rate bits/sec	Experiment Sampling Rate	Duration of Experiment	Distance from Earth km	Remarks
Ionospheric sounding and electromagnetic radiation	transmitter and receiver, antenna	40	200	1100	16	2/hr	2 mo	150×10^6	
Infrared photography	infrared vidicon, scanning device, preamplifier, 3 filters	9	20	500	30,000 bits (total)	1/ 3.5 hr	2 mo	150×10^6	8 shades of gray
Radar	high frequency transmitter and receiver, antenna	44	360	1200	16	2/hr	2 mo	150×10^6	
Photographs	camera, scan circuits, tape recorder, filters	26	20 (peak) 6 (av)	200	750,000 bits (total)	1/ 3.5 hr	2 mo	150×10^6	1.3×10^6 bits storage required 8 shades of gray
Cosmic ray	Cerenkov counter, ionization chamber, pulse height analyzer (8 ch), Geiger-Mueller tubes, high voltage P.S.	20	2	200	4	continuous	8 mo	150×10^6	8 levels
Mass spectrograph	collecting device (titanium band), mass spectrometer tube, logarithmic amplifier circuits	22	17	600	20	continuous	8 mo	150×10^6	
Magnetic field	magnetic resonance magnetometer	5	5	60	3	continuous	8 mo	150×10^6	requires nonmagnetic environment
Attitude sensor	photoelectric seeker	0.3	2	130	4	continuous	8 mo	150×10^6	
Timer		2.2	2.5 (peak) 1.2 (av)	60	—	continuous	8 mo	150×10^6	

Total weight of instrumentation .... 168.5 lb
Total weight plus structure .... 202.2 lb
Total power (if all experiments are sampling at the same time) . . 628.5 watts (peak)
Total volume .... 4050 in.³
Requirements: Injection guidance, attitude sensing and control, zero spin, terminal guidance, retro-rocket, 150×10^6 km telemetry, transmitter interrogation, maximum payload sterilization.

Table 15. Mars Entry (Payload No. 9, November 1962)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Information Rate bits/sec	Experiment Sampling Rate	Duration of Experiment	Distance from Earth km	Remarks
Atmosphere composition	collecting device (titanium band), mass spectrometer tube, logarithmic amplifier	20	17	800	100		1 sec	150×10^6 at impact	
Surface hardness	cone penetrometer, with impact transmitter	44	2	600	max. available BW		1 sec	150×10^6 at impact	
Temperature of atmosphere	thermocouple	0.1	0	1					
Photographs	camera, tape recorder, scan circuits, filter	9	20 (peak) 6 (av)	200	750,000 bits (total)	2/hr 10/sec		150×10^6 at impact	8 shades of gray
Communications	ruggedized beacon, antenna, power supply	60	600	1500					
Surface temperature	resistance thermometer	1	0.5	30					

Total weight of instrumentation . . . . 134.1 lb
Total weight plus structure . . . . 160.9 lb
Total power (if all experiments are sampling at the same time) . . 639.5 watts (peak)
Total volume of instruments . . . . 3131 in.^{3}
Requirements: Injection guidance, attitude sensing and control, zero spin, terminal guidance, 150×10^6 km telemetry, maximum sterilization.

Table 16. Lunar Soft Landing (Payload No. 11, June 1963)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.³	Total Information bits	Experiment Sampling Rate	Duration of Sample	Duration of Experiment	Remarks
Surface analysis and properties (spot samples)									
Visual survey (color and spectrum scan; panorama, close-up, and microscopic)	TV camera(s), microscope and telescopic lenses, filters, direction control, searchlight	33	40	700	1×10^6	at command	1 sec	2 weeks	8 shades of gray—complete control of direction, focus, etc.
Surface probe[a]	probe with thermistor in tip, vibration transducers, sonic generator, load train, and motion transducers	4	10	40	200	at command	5 min	2 weeks	
Temperature (surface and subsurface)[a]		—	0.2	—	10	at command	inst	2 weeks	
Thermal conductivity[a]		—	50	—	500	at command	15 min	2 weeks	
Sample collector (surface dust)[a]	either scoop, blower, or sticky probes (10); manipulators	10	25	120	—	at command	1 min	2 weeks	
Sample collector (subsurface rock, etc.)[a]	explosive side-wall samplers (5), drill, or pulverizer; manipulators	60	20	200	—	at command	1 min	2 weeks	
Radioactivity (U, Th, K, etc)[b]	scintillator and pulse-height analyzer, plus β source (scattering and X-ray fluorescence), γ source (scattering), separable Ra-Be source (neutron activation), and neutron detector.	30	30	500	1000	at command	30 min	2 weeks	sample collection optional; might be included in roving vehicle
Chemical analysis (X-ray fluorescence)[b]		4	30	10	3000	at command	15 min	2 weeks	
Chemical analysis (neutron activation)[b]		10	30	15	2000	at command	10 min	2 weeks	
Density of surface (β scattering)[b]		1	30	10	50	at command	1 min	2 weeks	
Density of subsurface (γ scattering)[b]		4	30	10	50	at command	1 min	2 weeks	
Magnetic material probe[b]	coil, oscillator circuit, amplifier	2	4	30	10	at command	inst	2 weeks	
Chemical analysis (X-ray fluorescence)[b]	sample manipulator & port, goniometer, Geiger counters, photo detectors, electron gun with target and filters, light sources, (visible, ultraviolet and infrared); scanning crystal(s) and gratings, sample heater, and ultraviolet excitation source	35	50	1200	3000	at command	15 min	2 weeks	sample collection and manipulation probably necessary—might also be included in roving vehicle
Phase analysis (X-ray diffraction)[b]		2	50	—	5000	at command	15 min	2 weeks	
Phase analysis (infrared luminescence)[b]		2	10	—	200	at command	5 min	2 weeks	
Reflection spectra[b]		2	10	—	1000	at command	5 min	2 weeks	
Thermoluminescence[b]		2	20	—	1000	at command	10 min	2 weeks	
Chemical analysis (ultraviolet emission spectra)[b]		4	30	—	5000	at command	2 min	2 weeks	

Table 16 (Cont'd)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Total Information bits	Experiment Sampling Rate	Duration of Sample	Duration of Experiment	Remarks
Volatile materials; also atmosphere[b]	sample collectors, heaters, mass spectrometer	25	30	700	1000	at command		2 weeks	requires sample manipulation (tentative)
Specific heat and thermal analysis[b]	sample chamber, heater, thermocouple, amplifier	5	25	40	500	at command	20 min	2 weeks	requires careful sample packing
Sublife or organic matter[b]	undecided; may have to be inferred from other experiments	4	1	122	1000	at command	—	2 weeks	tentative
Environmental measurements									
Seismic activity	seismograph	30	2	300	2/sec	continuous	—	>1 month	could study sound propagation by setting off explosive charge
Magnetic field (intensity and direction)	electron spin resonance magnetometer	4	5	60	3/sec	continuous	—	>1 month	
Gravity	torsion gravitometer (fibre type, ruggedized)	1	1	30	10/sec	continuous	—	>1 month	requires temperature calibration
Temperature (radiant, surface, and subsurface)	same as surface probe, plus additional thermistors		0.2	40	10/sec	continuous	—	>1 month	
Solar radiation (rf to γ rays)[b]	modification of other experiments (visual survey, radioactivity, etc)								single instrument unfeasible, best inferred from other experiments
Return of samples (up to 1 lb)[b]	rockets (5)	200	10	3300	—	at command	—	2 weeks	
	rocket launcher	75	10	1300	—	at command	—	2 weeks	
	orientation equipment	45	50	750	—	at command	—	2 weeks	
Roving vehicle (containing sample collectors, etc.)	vehicle, motors, steering gear,	100	200	1500	—	at command	—	2 weeks	
Program control and automation equipment[b]	command receiver, logic circuits, tape recorder(s), timer(s)	55	50	1000	—	—	—	2 weeks	

Total weight of instrumentation .... 749 lb
Total weight plus structure .... 899 lb
Total power .... 853.4 watts
Total volume .... 11,977 in.^{3}

[a]Must be included in roving vehicle.
[b]Desirable to include in roving vehicle, if space and power permit.

The low gravitational field of the moon makes it appear feasible to use small rockets to send samples of the lunar surface back to earth (although the problem of locating and recovering the returned sample is formidable) for detailed chemical and petrographic analysis. It should be noted, however, that such data, on isolated samples, would only have great value in the context of an area survey, and therefore a sample-return complements, but does not supplant, the *in situ* experiments.

From a scientific standpoint, the ideal configuration would utilize a completely self-sufficient roving vehicle, containing all of the surface exploration equipment together with the command receiver, program control, logic circuits, power supply, and transmitter and antenna for telemetering the data back to earth. Using this configuration, the roving vehicle would have a hypothetically unlimited exploration range, which might greatly increase the value of the data. Sample collection could be simplified or even eliminated for most experiments. On the other hand, such a vehicle would be quite large and would require a powerful motive system, the largest source of power consumption in the whole payload. Moreover, since the operation of any mobile vehicle will be somewhat risky and subject to unforeseeable hazards, it will place the experimental package in jeopardy.

The stationary package is by comparison quite small and low in power consumption. A seismograph, magnetometer, gravitometer, and temperature probe could operate continuously on 25 watts. It therefore seems feasible to relay this data via the moving vehicle or, better, to telemeter it direct to earth by a small, narrow-bandwidth transmitter.

Another configuration might involve a much smaller roving vehicle containing only camera, sample collectors, and surface-texture probe. This would probably be connected by power and communication cables to the main payload package which would contain the sample receivers, manipulators and all of the experimental instruments. The main payload package might also contain a few sample collectors, so that in the event of loss or failure of the roving vehicle, some data could be obtained. Such a configuration would effect some saving in power supply and perhaps also in structure weight. It would involve less of a risk of locomotion failure but, on the other hand, it would necessitate elaborate sample manipulation. Moreover, the exploration range would be limited by the cable.

If sample collection and manipulation involves insuperable problems, it will be necessary to resort to the larger roving vehicle containing no sampling experiments. Conversely, if vehicle locomotion is unfeasible or very risky, a completely stationary payload may be necessary. (In the latter event, the sample-return rockets would become even more important.) The choice will therefore depend not only on the available payload weights and power supplies, but on the satisfactory development of a reliable mobile vehicle and sample collection system and also on the surface texture and terrain data obtained from the lunar rough-landing probe.

12. Venus soft landing (Payload No. 12, March 1964). The mobile surface-exploration vehicle, proposed for the lunar soft landing, constitutes only part of a satisfactory exploration program for Venus or Mars. A thorough investigation of the weather and atmospheric conditions is of even greater importance, and the question of life on other planets is of such universal concern that some sort of biological experiment, however rudimentary, should be included in the first soft landing.

In addition to the usual weather data—pressure, temperature, humidity, wind direction and velocity—the stationary instrument package should provide data on the Venusian daylight spectrum, and its variation with direction and time. Atmospheric composition can be determined by a mass spectrometer and long light-path spectrophotometer. The latter will also give some data on the presence of dust, fog, etc. A microphone will detect atmospheric noises. In addition, a focused sounder in conjunction with a microphone on the moving vehicle will give data on atmospheric sound propagation. The stationary structure would also include a seismograph, magnetometer, gravitometer, and soil probe.

The stationary structure will have provision for launching radiosonde balloons so as to obtain high-altitude weather data. In addition, most of the stationary instruments in the stationary package can be designed to function during entry.

The surface-exploration vehicle will contain, in addition to the all-important color television camera, a small radar scanner so that some topographic data can be obtained even if there is atmospheric interference with photography. The remaining experiments will be similar to those of the lunar soft-landing vehicle. However, since the Venusian surface will be far more unpredictable than the lunar surface, it seems expedient to avoid experiments involving sample collection and manipulation. The sole exception would be a suction pump and filter for the collection of airborne solid particles; this has very high priority, since if any form of life were present, it would probably include airborne microorganisms. This sampler would be used in conjunction with a special microscope (also usable for soil and rock examination) and some simple biological experiment, as yet unspecified.

As with the lunar soft landing, the mobile vehicle should be capable of being directed to optimum sampling spots by command from earth, these commands being decided upon on the basis of the photographic data.

The communication and telemetering system is as yet undetermined. Radio communication through the Venusian atmosphere may be poor enough to warrant telemetering data from the surface-exploration experiments to a simultaneously launched satellite containing a powerful transmitter for relay to earth.

The experimental program proposed here is admittedly ambitious, not only because it involves a complex network of instruments, logic circuits, and program controls, but because it presupposes that the earlier satellites and entry probes will send back a maximum amount of reliable information and that this information will be favorable, i.e., that an atmospheric radio "window" exists, that photographic data has been obtained and discloses suitable soft-landing sites, etc. Should the data be ambiguous or unfavorable, it would be best to simplify the payload by incorporating the mobile-vehicle experiments into the stationary structure, though this would still probably require a smaller, sample collecting, roving vehicle.

It should be borne in mind, however, that the optimum planetary payload (i.e., with mobile vehicle) is readily adaptable to a Mars soft landing and that an opportunity for a Mars soft-landing shot occurs a few months after the Venus soft landing proposed here. Our present knowledge indicates that a Mars soft landing would have few of the environmental problems encountered on Venus or the moon, and would have a considerably greater chance of success. Therefore, during the development of the lunar and planetary soft-landing payloads, the possibility of a Mars landing should always be kept in mind and, if necessary, given preference.

There is some doubt as to the nature of the surface of Venus. In the event that the entry probe indicates a water surface, a special water-landing payload could be developed, quite rapidly and easily, which would have a high probability of giving ample useful data. If, at the time of the Venus soft landing, there is still some uncertainty as to the existence of water-covered areas, it may be possible to partially ensure the regular soft-landing payload (as described above) against this hazard. The radiosonde launcher would contain a sufficient gas supply to inflate pontoons adequate for floating the entire payload; however, the complications of waterproofing, orientation, etc., may make this unfeasible.

The Venus-entry payload (Payload No. 7) and this Venus-landing payload presents serious communication problems. In both cases, it cannot be considered feasible to bring down a large parabolic antenna through the atmosphere of Venus. Since the information to be transmitted to earth from the simpler Venus-entry payload does not require a large bandwidth, that earlier payload might be able to use a simple dipole as a transmitting antenna after it arrives at the surface of Venus. But this situation does not hold for the Venus-landing package.

For the Venus-landing package, large volumes of information will be collected, including picture-type information. Thus, a large communication bandwidth is required. This, in turn, implies the use of a comparatively large parabolic reflector for the transmitter antenna.

Two solutions suggest themselves for accomplishing this requirement: (1) a foldable antenna could be included with the stationary instrument package. After a safe landing on the surface of Venus, this antenna would be deployed and swiveled to aim at earth. If the planet rotates, the antenna would have to track the earth in a manner consistent with the rotation rate of Venus. Such an antenna would have to be strong enough to withstand possible surface winds and rigid enough to support itself against the Venusian gravity. Clearly a large amount of the payload weight would be devoted to such an antenna if this solution were adopted. (2) The total payload injected into the Venusian orbit from earth might consist of two separate pieces which could be separated from each other after the final course-correction guidance maneuver prior to arrival at the target planet. One package, the actual landing package, would be provided with an additional course-correction rocket to direct it into the atmosphere of Venus. The portion left behind would follow the trajectory previously established to a point 1000 miles or so above the surface of Venus, where it would, with the help of a retro-rocket, become a satellite

Table 17. Venus Soft Landing (Payload No. 12, March 1964)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Total Information bits	Experiment Sampling Rate	Duration of Sample	Duration of Experiment	Remarks
Surface analysis and properties (spot sampling)									
Visual survey (color spectrum, scanning, panorama, close up)	TV camera(s), microscopes, telescopic lenses, filters, direction control, search light	33	80	700	1×10^6	at command	inst	2 weeks	8 shades of gray, complete control of direction, focus, etc.
Radar survey	transmitter, antenna, receiver, direction control	44	50	940	1×10^5	at command	10 sec	2 weeks	
Surface texture probe	probe, vibrator, amplifier, transducers, thermistor, hygrometric resistor in tip	4	10	38	200	at command	5 min	2 weeks	may require two probes: one in roving vehicle, one in stationary payload
Soil temperature		—	0.1	—	10	at command	1 sec	2 weeks	
Soil humidity		—	0.1	—	8	at command	1 sec	2 weeks	
Radioactivity (U, Th, K, etc.)	scintillator and pulse-height analyzer, plus β source (scattering and X-ray fluorescence), γ source (scattering), separable Ra-Be source (neutron activation) and neutron detector	30	30	500	1000	at command	90 min	2 weeks	scintillator pressed against surface to prevent atmosphere scattering
Chemical analysis (X-ray fluorescence)		4	30	10	3000	at command	15 min	2 weeks	
Chemical analysis (neutron activation)		10	30	15	2000	at command	10 min	2 weeks	
Density of surface (β scattering)		1	30	10	50	at command	1 min	2 weeks	
Density of subsurface (γ scattering)		4	30	10	50	at command	1 min	2 weeks	
Infrared luminescence	sealed tube with sapphire windows, light sources, grating of filters, detectors, amplifier	11	20	310	200	at command	5 min	2 weeks	sealed light path tube, pressed against surface, prevents atmospheric absorption
Reflection spectra (ultra-violet to infrared)		—	20	—	1000	at command	5 min	2 weeks	
Microscope	camera, microscope, filters, manipulator, electronics	8	40	175	1×10^6	at command	inst	2 weeks	8 shades of gray, focused on surface
Airborne particle sampler	suction pump, milipore dish, manipulators	7	20	125	—	at command	at command	2 weeks	used with microscope and scintillator
Biological experiment	possibly culture tubes	5	—	100	500	at command	at command	2 weeks	used with microscope and airborne sampler
Atmospheric and environmental measurements									
Pressure (<1 micron)	ionization gauges (3)	3	10	5	—	entry only	5 hr	—	designed to provide additional data during entry
Pressure (<1 mm)	Pirani gauges (3)	0.1	5	2	—	entry only	5 hr	—	
Pressure (>1 mm)	bellows (2), and transducers	0.5	0.5	5	10	1/hr	1 sec	2 mo	
Wind (direction and velocity)	pitot tube rosette, bellows, transducers	4.5	0.2	8	20	1/hr	1 sec	2 mo	

Table 17 Cont'd)

Experiment	Instrumentation	Weight lb	Power watts	Volume in.^{3}	Total Information bits	Experiment Sampling Rate	Duration of Sample	Duration of Experiment	Remarks
Air temperature	thermistor	0.1	0.1	1	8	1/hr	1 sec	2 mo	
Air humidity	hygroscopic resistor	0.1	0.1	1	5	1/hr	1 sec	2 mo	pressure calibration necessary
Air composition	mass spectrometer, with ballast chamber and pump	20	20	700	1000	1/hr	1 min	2 mo	should also function during entry
"Daylight" radiation spectrum	photocell cluster, gratings or spectral filters, amplifier, mirror and light source (for absorption spectra)	6	10	60	1000	1/hr	1 min	2 mo	directional
Air absorption spectrum		1	20	50	5000	1/hr	5 min	2 mo	
Magnetic field (directional)	alkali vapor resonance magnetometer	4	5	60	20	1/hr	1 sec	2 mo	
Gravity	torsion fibre gravitometer	1	1	10	10	1/hr	1 min	2 mo	
Seismic activity	seismograph	30	2	500	10/sec	continuous	continuous	2 mo	
Atmospheric sounds and sound propagation	microphones (2), amplifier, sounder (focusing)	4	5	60	5000/sec	intermittent	5 min	2 mo	
High-altitude pressure, temperature, humidity	radiosondes (10), phototube, balloon, transmitter, battery, bellows, thermistor, hygrometer	60	—	2000	50	3/min	10 sec	0.5 hr/radiosonde	
	radiosonde launcher, gas supply	30	50	500	—	at command			balloon inflation command controlled
High-altitude winds	radiosonde tracker, ranging and tracking receiver, antenna and erector	50	100	800	30	3/min	10 sec	0.5 hr/radiosonde	
Direction orientation	gyro-compass	2	5	50	—	at command	—	—	
Special structure	environmental adaptation of weather station		—	—	—	—	—	—	
Program control of data handling	command receiver, logic circuits, tape recorder, timer, etc.	55	35	1500	—	at command	—	—	
Roving vehicle	alternate configurations: containing all "surface analysis and properties" experiments with provision for rudimentary subsurface sampling; containing only camera and/or radar scanning, surface texture probe, and sample collecting devices	200	500	2000	—	at command	—	—	

Total weight of instrumentation .... 632.3 lb
Total weight plus structure .... 758.8 lb
Total power .... 1159.1 watts
Total volume of instruments .... 11245 in.^{3}

of that planet. The landing package would be equipped with a parabolic reflecting antenna having a diameter small enough so that it could be enclosed within the protective housing of the landing package during entry through the atmosphere and simply erected after arrival on the planet's surface. This antenna would serve to communicate the results of the experimental findings on Venus to a receiver located in the orbiting portion of the payload. This Venus satellite would then act as a relay station equipped with necessary antennas for receiving the information from the surface and transmitting it back to earth. If this solution were employed, a large fraction of the gross payload weight would have to be devoted to the satellite portion of the dual payload.

The answer as to which of these two solutions is the most feasible, or whether or not still another solution exists, must await a more thorough design study.

The characteristics of the Venus soft-landing probe are listed in Table 17.

C. Development Schedule

1. Procurement of engineering design data. The engineering design of a space probe is a task which draws upon knowledge at the very frontiers of many branches of technology. It is a difficult job even if the entire environment with which the probe has to contend in order to accomplish its mission is well known. There are, in fact, many aspects of the space environment about which little or no information exists. A good example is our lack of knowledge of the type of surface and atmospheric environment in which a Venus soft-landing probe would be required to operate. For these reasons any rational, long-range space-exploration program must be a carefully planned, step-by-step procedure in which engineering design data for the later vehicles is obtained as fully as possible by earlier vehicles. If the over-all program is to be a coherent one, the earlier vehicles must therefore be instrumented with this requirement in mind. If the eventual goal is to put a man into deep space, it is obvious that the necessity for accurate information concerning the environment he must face is critical.

As an example of the type of knowledge of the space environment that is necessary for engineering design purposes, the problem of temperature control should be considered. The instrumentation in a space probe operates efficiently only over a restricted temperature range, which makes it necessary to hold the temperature within fixed limits. Considerable experience in this area has been gained from the satellite and lunar vehicles that have already been launched. The temperature-control methods hinge on control of surface emissivities. The long-term effects on these emissivities of such space conditions as vacuum, micrometeorite impacts, corpuscular radiation, gamma radiation, and ultraviolet radiation are not well known. Two things are obviously necessary in order to acquire this needed information efficiently. The first is to determine what the characteristics of the radiation, vacuum and micrometeorite environment are, and the second is to develop ground testing facilities for simulating this environment, whenever possible. The success of the temperature-control system must then be put to the final test by monitoring and telemetering back temperatures at various points in the vehicle during its actual flight.

In organizing the projected NASA program, considerable attention has been given to the preceding requirements. The early interplanetary shots devote much of their instrumentation to the measurement of such space environmental conditions as micrometeorite erosion and cosmic radiation. Meteor detectors might be included to determine the meteor hazard to manned probes. The planetary satellites and entry probes are equipped to obtain atmosphere and surface data for use in designing soft-landing probes.

If an accurate measurement of an important unknown quantity is planned for a certain shot, it is well to obtain at least a rough idea of the value of this quantity from an earlier shot. (This at least determines the order of magnitude of the quantity and makes it unnecessary to design the instrumentation to measure over a range of several orders of magnitudes. Measuring over a large range often involves the use of logarithmic amplifiers with their associated inaccuracies.) Whenever possible, this procedure has been applied in the present program.

2. Ground test requirements. The ground testing of payloads associated with this program must be more extensive and thorough than test programs employed in the development of a missile weapons system, since the number of vehicles and shots involved is severely limited.

Some of the difficult testing problems which must be solved as soon as possible are listed as follows:

1. In the past most airborne equipments have been required to operate for relatively short periods—minutes, hours, a few days, or a few weeks. Many of the probes considered here require satisfactory operation for periods of several months. Present schedules do not permit realistic life tests; therefore it will be necessary to establish meaningful accelerated life tests on critical components and component parts.
2. There is relatively little information available regarding the effects of hard vacuum with infinite pumping speeds on conventional construction materials. It does not appear that vacuum chambers of sufficient size to accept an entire payload and to simulate spatial vacuums will be available in time for testing the escape-toward-Mars and escape-toward-Venus probes.
3. The evaluation of attitude control and sensing systems in a gravity-free environment poses special problems as yet unsolved.
4. The effects of radiation on certain classes of component parts have not been determined. Radiation-simulation test facilities should be made available as soon as possible.
5. One of the major problems associated with the design of deep-space probes is the control of payload temperature. There is need for facilities to accurately determine the reflectance characteristics of proposed external surfaces and materials. Further, there is need for a vacuum facility which incorporates radiation sources and sinks which will to some degree verify expected thermal time constants and payload operating temperatures.

Considerable work has been done in establishing test equipments and procedures for the environmental simulation of: vibration, shock, high and low temperature, humidity, sand and dust, linear or static acceleration, and spin.

The capabilities of present test equipments will probably be adequate for testing the early probes. However, it is reasonably certain that some equipment capabilities (vibration, shock, linear acceleration) will have to be doubled or tripled as larger payloads are considered.

It is important that environmental specifications be established early in any development program. Every effort should be made to provide adequate instrumentation for the determination of vibration and shock environments based on static firings of proposed propulsion systems. Specifications based upon inadequate information result in inadequate or over-designed components.

3. Typical schedules. The development of payloads for lunar and planetary exploration is a problem of greater complexity than the development of the missile systems which have been carried out to date. The development schedules reflect this fact. In particular, the requirement for a long reliable lifetime implies the necessity for a long environmental testing program preceding the actual launching. It is not likely that this testing period can be made a great deal longer than the actual flight time for some of the planetary missions, and in some cases it may be difficult to make it even equally long. Thus, if any component fails during this life test and indicates the necessity for a redesign, the redesigned version cannot have a life test of really adequate duration. This fact implies a great emphasis on the need for reliable subcomponents and elements. It would not be advisable to introduce a new type of transistor, for example, into the payload design at a late stage in the development program. It would be much more desirable to rely only on transistors which have been in use for a long time before their incorporation into the planetary payload, transistors for which a large background of reliability experience is available.

This conclusion implies that innovations in elements and component design cannot be readily introduced into the program. Although this may place a limitation on the versatility and capability of the payloads, it is a necessity in order to assure any degree of confidence in the final success of the mission. Another fact which makes the payload development problem more complex than the development of a missile system is the rigidity of the flight schedule. If, because of development difficulties, the flight schedule has to be delayed for as much as one week, then actually the flight time will be delayed for a year and a half to two years. Still another fact which adds to developmental difficulties is that each of the payloads developed for the planetary exploration program is different from its predecessor, in some cases to a very large degree.

All of these developmental problems are reflected in the development schedule. An example of a development schedule is given in Fig. 64. This is the schedule for the

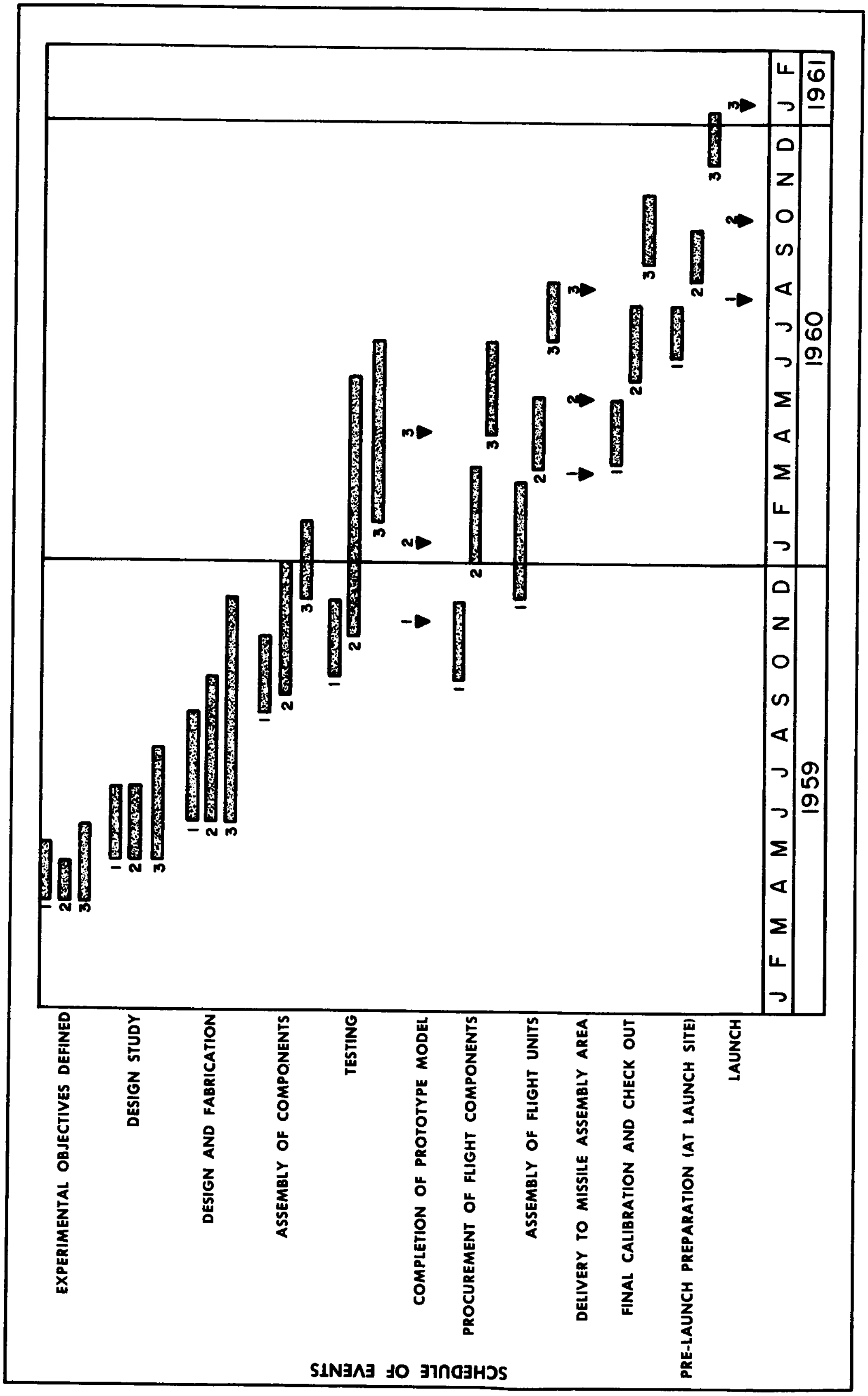

Fig 64. Typical Development Schedule, Payloads No. 1, 2, and 3

development of the first three payloads listed in the suggested program.

Some elements of this schedule are worth special attention. First, notice that the experimental objectives for Payload No. 2, listed as an escape toward Mars, must be defined by the end of April 1959. Furthermore, for this same payload, it should be noted that component testing must begin November 1959 and continue through the month of May 1960. The prototype model of this payload will be available in mid-January of 1960. Thus, the life testing of the prototype model can be carried out only for four and one half months, whereas the actual flight time of this payload will be at least five months, or longer.

The development program for the Mars escape payload is further complicated by the fact that the Lunar-Miss payload (Payload No. 1) and the Escape-Toward-Venus payload (Payload No. 3) must also be designed and developed during the same time intervals as those for the Escape-Toward-Mars payload. This duplication of effort can be carried out successfully only if all three payloads are basically similar.

The development schedule for the Mars payload allows 17 months between the definition of the experimental objectives and the launch date. Although this may seem like quite an adequate time span, inspection of the various portions of the schedule will show that actually the time allowed is quite short.

An alternate schedule for the development of such a payload is given in Fig. 65. This schedule has been constructed with maximum emphasis on reliability of the final payload, and is thus very conservative. From the point of view of the development engineer, however, this conservative schedule is quite realistic, and desirable. Unfortunately, this schedule calls for nearly 4 years between the definition of experimental objectives (which would have had to occur at the end of January in 1957) and the launch of the final payload in October of 1960.

A realistic schedule which permits the incorporation of new ideas and information which might be gained by other flights in the planetary exploration program, but also permits adequate testing for the assurance of reliable operation, would lie somewhere between the comparatively tight schedule shown in Fig. 64 and the very conservative schedule shown in Fig. 65.

As the payloads become more complex, laboratory environmental testing will not be adequate to assure the successful operation of the payload system. Flight tests of the complete system will be required. Figure 66 shows a proposed schedule of major payload system tests associated with the development of several of the payloads listed in the proposed program. These system tests include air drops to check out the behavior of payload systems destined for planetary landings during the final phases of their landing and surface operation. Tests in sounding rockets as well as tests in earth satellites are indicated for payloads which must behave as satellites of other planets.

For payloads designed to land on the surface of the moon, high-speed-impact tests are shown. It is the function of these tests to launch the payload toward the earth

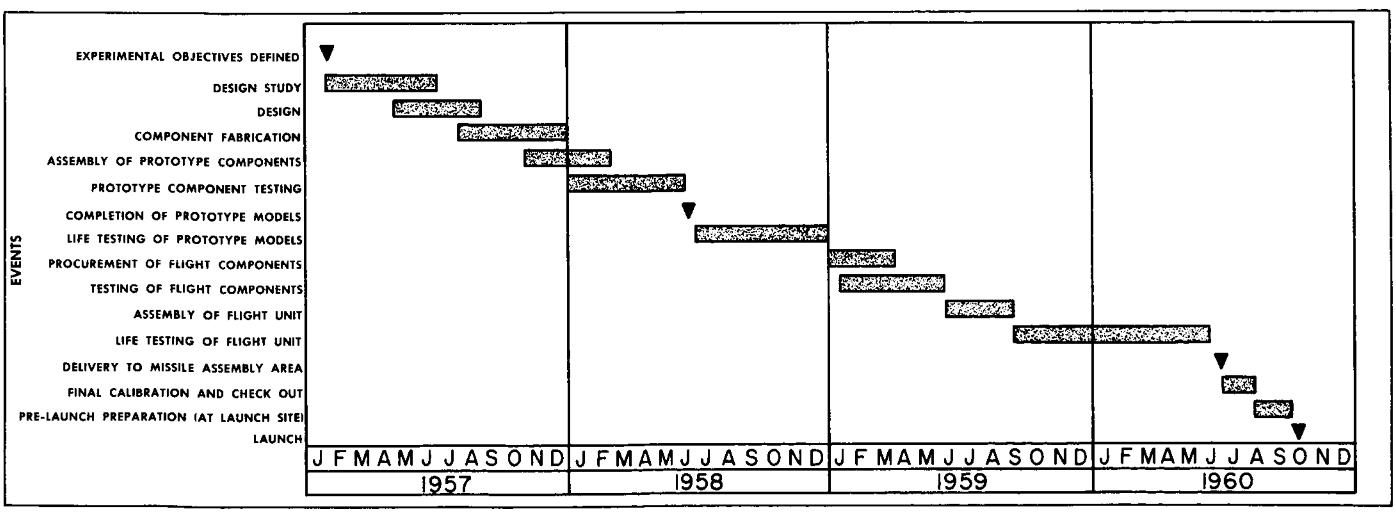

Fig. 65. Conservative Development Schedule, Payload No. 2

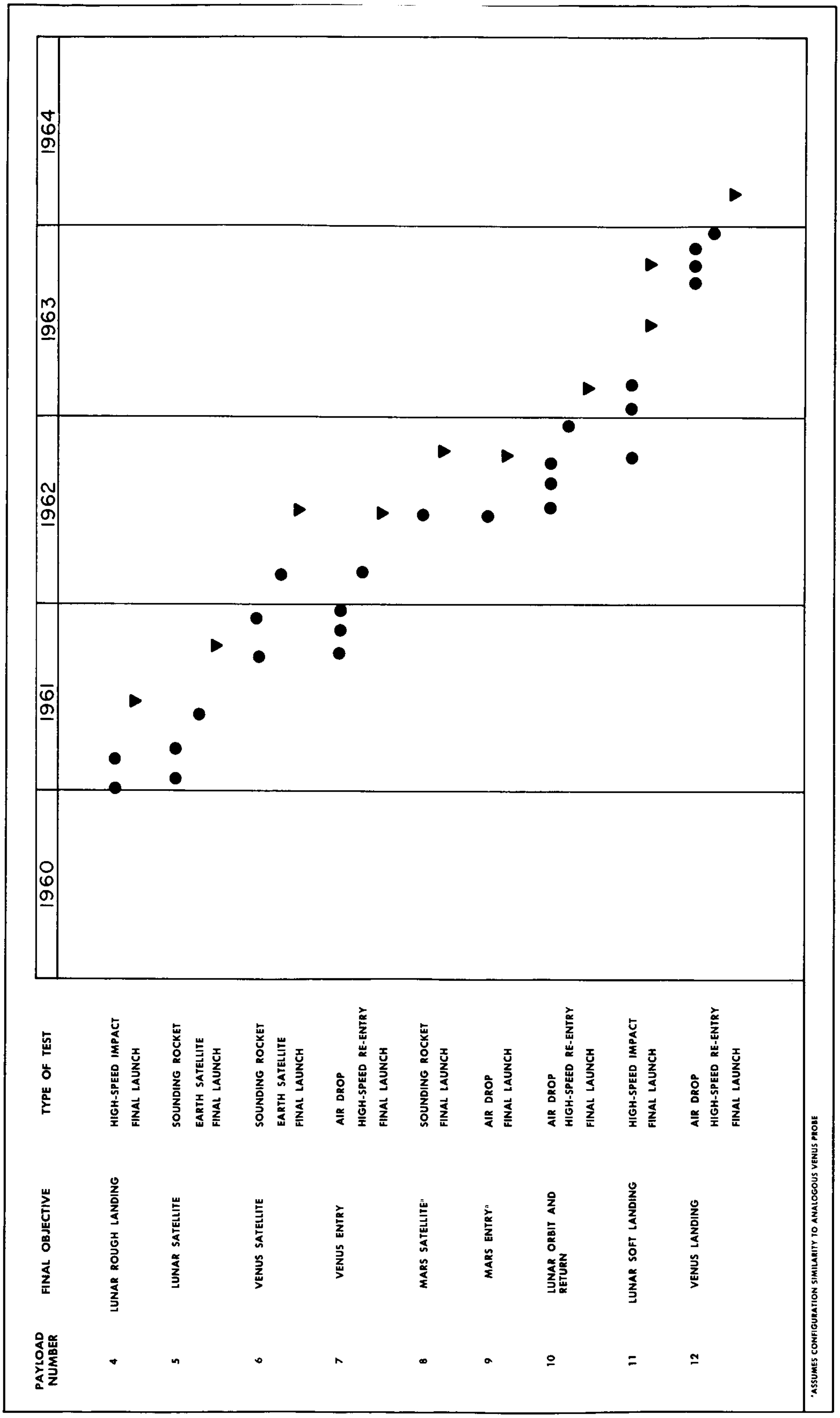

Fig. 66. Suggested Schedule of Major Payload System Tests

at speeds comparable to those which will be attained by the payload on its approach to the moon. Thereafter, the retro-rockets will be ignited to slow the payload down for a landing on the surface of the earth. The function of the payload after the landing will also be checked.

For payloads destined to enter the atmosphere of other planets, the preliminary air-drop tests will be followed by high-speed re-entry tests into the earth's atmosphere. The speeds obtained during these tests must be in excess of the escape speed of the particular planetary target. For Venus, the required re-entry test speed is between two or three times the speed attained by an ICBM on re-entry into the earth's atmosphere.

To meet the schedule for such system tests, a complete prototype payload must be available in time for the earliest system test. In some cases, this is several months before the actual launch date. For example, for the lunar soft landings, high-speed-impact tests should begin late in 1962, although the actual launching of this payload does not occur until mid-1963.

The lead times necessary to meet the development and testing schedules for lunar and planetary payloads may seem unnecessarily long to those outside of the missile development industry. However, such lead times must be allowed if the program is to meet with any degree of success.

It is in the area of scheduling that the effect of public reaction makes itself felt most keenly. The public demand for dramatic "firsts" overrides the public concern about test failures. It is in response to this public demand that development schedules cannot be made as long as might be desirable if reliability were the sole objective.

On the other hand, neither the public nor the scientists nor the engineers would be satisfied with a program consisting only of a monotonous series of failures. Somewhere between the two extremes, long developmental testing on one hand and a completely crash program on the other, lies the desirable scheduling philosophy. The schedule presented in Fig. 64 for the development of the first 3 payloads in the series is perhaps a realistic picture of such a middle course for these comparatively simple payloads.

VII. CONCLUSIONS

The development of a typical payload considered in this Report is in many ways analogous to the development of a complete guided-missile system. The payloads contain guidance and control devices, communication devices, telemetry equipment and measuring devices, and in many cases a major portion of the payload weight is devoted to a rocket propulsion system.

Actually, the development problems associated with these exploration payloads are much more difficult than the development programs associated with the typical guided missile. Each of these payloads will be different from its predecessor both in weight, in objectives, and in component design. Furthermore, each payload must be capable of operating without the benefit of last minute checkout or adjustments, after having travelled for days, or perhaps months, through the vacuum of space. Some of the payloads must successfully enter an atmosphere of only partially known characteristics traveling at a speed of more than twice that attained by an ICBM on re-entry into the earth's atmosphere. Their "warhead" will not be a single device designed to operate only once, but rather a whole array of devices, some of which may operate only once, some of which will operate continuously, and others periodically at intermittent intervals.

Many of the mechanical devices must automatically carry out a program of exploration and analysis without the help of human maintenance or human direction. In many cases, they must perform this test in a completely unknown environment, for it will be their job to discover the properties of this new environment. If these devices should fail to operate properly, it will be difficult, if not impossible, to detect the cause of their failure and so make the necessary design changes before the next attempt.

At the present time, we have no technology which would permit the design and construction of automatic devices to be carefully carried out to some portion of the earth's surface, and then left to perform their function of analyzing the characteristics of the earth. And yet, in a few short years, we must design such devices to operate on the moon and the near planets.

The public demands sudden and spectacular achievement in their space program. This demand cannot be ignored nor relegated automatically to second place in comparison with the demands of reliability. Both demands must be met.

The development of the payloads for the exploration of space is a task which will test the limits of our ingenuity; but most of all, it is a task which must begin immediately if it is to have any hope of success within the proposed time scale.